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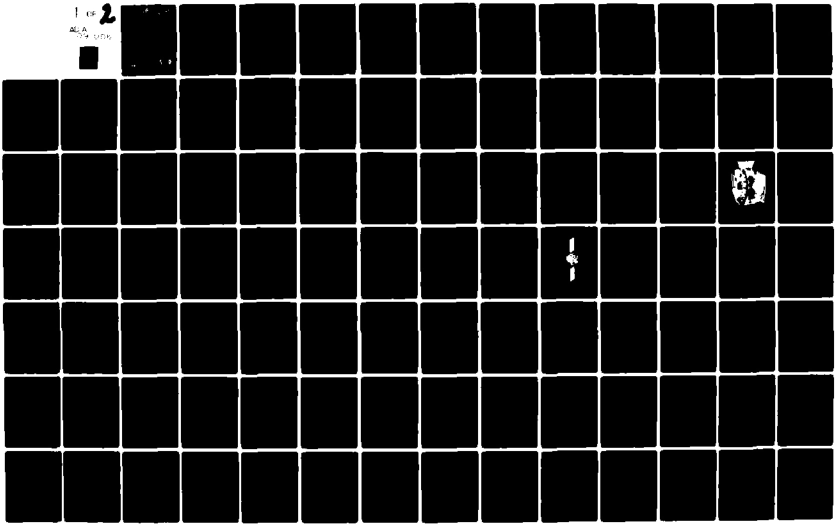

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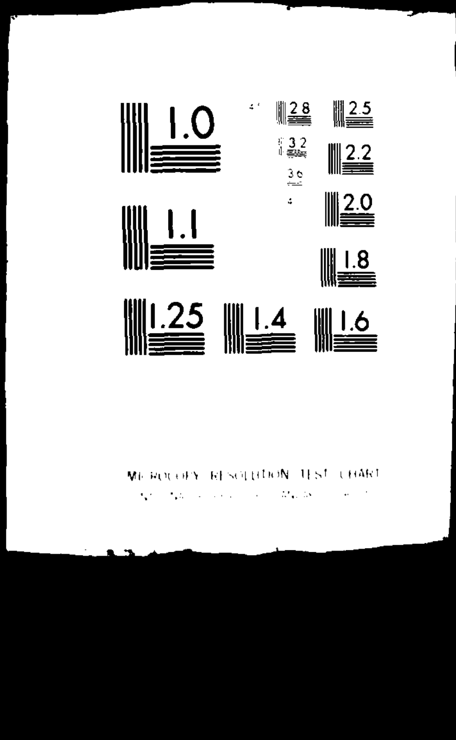

LEVEL II

AFRPL-TR-80-38

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PULSED PLASMA PROPULSION SYSTEM/SPACECRAFT DESIGN GUIDE

SEPTEMBER 1980

AIR FORCE ROCKET PROPULSION LABORATORY
DIRECTOR OF SCIENCE AND TECHNOLOGY
AIR FORCE SYSTEMS COMMAND
EDWARDS AFB, CALIFORNIA 93523

TRW
DEFENSE AND SPACE SYSTEMS GROUP
ONE SPACE PARK
REDONDO BEACH • CALIFORNIA

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FOREWORD

The purpose of this publication is to describe the principle characteristics of the millipound pulsed plasma propulsion system as an aid to potential spacecraft designers and users of electic propulsion. It contains performance and interface data, design constraints, and various other considerations applicable to the implementation of the pulsed plasma propulsion system.

Additional information supplementing the data presented herein may be obtained from the Air Force Rocket Propulsion Laboratory, Edwards Air Force Base (805-277-5540) or the Propulsion and Combustion Systems Department at TRW Defense and Space Systems Group (213-535-2850).

Robert J. Vondra

ROBERT J. VONDRA
Project Manager

Walter A. Detjen

WALTER A. DETJEN
Ch, Satellite Propulsion Branch

FOR THE COMMANDER

BEN A. LOVING, Lt Col, USAF
Chief, Liquid Rocket Division

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The purpose of this publication is to describe the principle characteristics of the millipound pulsed plasma propulsion system as an aid to potential spacecraft designers and users of electric propulsion. It contains performance and interface data, design constraints, and various other considerations applicable to the implementation of the pulsed plasma propulsion system.

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1. INTRODUCTION

The millipound, solid Teflon,* pulsed plasma propulsion subsystem has evolved over a period of 9 years from basic experiments demonstrating feasibility to the fabrication of a total system including power conditioning, control, and propellant supply. This design guide describes the hardware that has been developed, contains technical data pertinent to its integration on a spacecraft, and gives examples of typical spacecraft that could be designed to use the equipment.

The Air Force Rocket Propulsion Laboratory (AFRPL) is currently pursuing the necessary background technology to build a flight-qualified solid Teflon pulsed plasma thruster subsystem with a thrust of 1 millipound at 2200 seconds specific impulse. Fairchild Republic Company carried out major portions of the development work. TRW Defense and Space Systems Group has studied application and integration of the thruster subsystem and has prepared this design guide for the AFRPL.

1.1 ADVANTAGES OF PULSED PLASMA PROPULSION

The high specific impulse of the pulsed plasma thruster enables significant weight savings to be achieved when compared with existing propulsion subsystems. When pulsed plasma propulsion is used on 2000 pound military communications satellites, about 570 pounds additional weight margin is afforded for a dual launch with Shuttle-IUS. Such margin can compensate for some launch vehicle shortfall, permit additional communications equipment on board, or be used for increased component redundancy.

The pulsed plasma thruster provides extremely low impulse bits at repetition rates up to 1 pulse every 5 seconds. Impulse bits are 5 millipound-seconds each. Thus, the thrusters can be used for fine limit-cycle attitude control. On spinning satellites, they can be used in place of pulsed hydrazine thrusters for on-orbit operations. On three-axis stabilized spacecraft, it has been shown that pulsed plasma thrusters can be used as the principal torquing devices of the attitude control subsystem for on-orbit normal mode control, thereby replacing the reaction wheels.

*Trade name, DuPont

The fact that the thruster uses a solid propellant eliminates the need for tankage, feed lines, seals, or valves, and makes it readily compatible with a space environment. It may be used both on spin-stabilized and three-axis stabilized spacecraft.

1.2 MILLIPOUND THRUSTER APPLICATIONS

The primary use for pulsed plasma thrusters is for ΔV thrusting where the total impulse required is of sufficient magnitude to provide an opportunity for significant propellant weight savings compared to conventional propulsion. Since the average thrust level is low, the long mission life is needed to produce the high impulse. Higher orbit missions tend to be favored over lower orbits because of lower thrust level requirements and because propellant weight savings have a greater cost impact. The low disturbance torques associated with pulsed plasma thrust levels minimize their impact on the attitude control system.

Once pulsed plasma thrusters have been selected for primary ΔV functions, they are naturally available for performing other propulsive functions as well. For example, their repeatable small impulse bits make them well suited for stable limit cycle attitude control. Their small impulse bits are also well suited for precision low impulse orbit control.

Several applications for the millipound thruster are discussed in more detail in Section 3.5 of this design guide. Specifically, these are auxiliary propulsion for a geosynchronous communication satellite, geosynchronous stationkeeping of a spinning surveillance satellite, and orbit control for a 12-hour orbit navigational satellite.

1.3 PRINCIPLES OF OPERATION

The pulsed plasma thruster uses a burst of electrical energy to produce, accelerate, and eject a plasma wave.[1, 2] It differs from most

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[1] S. G. Rosen, "Colloid and Pulsed Plasma Thrusters for Spacecraft Propulsion," AIAA 73-1254, November 1973.

[2] B. A. Free, W. J. Guman, B. G. Herron, and S. Zafran, "Electric Propulsion for Communications Satellites," AIAA 78-537, April 1978.

propulsive devices in that it inherently produces discrete impulse bits. The same system, however, can also be used to produce the equivalent of a sustained steady-state thrust whose amplitude is directly proportional to the thruster's pulsing frequency.

The solid propellant pulsed plasma concept is illustrated in Figure 1. A set of rail-shaped electrodes is connected directly to a capacitor. A space between the electrodes allows the propellant to enter the electrode nozzle and rest against a fuel-retaining shoulder. The Teflon propellant is held against this shoulder by a constant force spring independent of thruster attitude, thermal environment, absence of gravity, satellite spin rate, or duration of thruster operation. A solid-state igniter plug is located in the cathode electrode. Power for the thruster is provided by an electronics package which provides energy for the capacitor and controls ignition in response to a command signal.

To generate an impulse bit, the power conditioner is commanded to charge the thruster capacitor to its designated operating voltage. This voltage also appears simultaneously across the space between the electrodes bounded by the propellant. Since the vacuum of space cannot sustain an electrical discharge, the applied voltage is retained until a simple discharge initiating circuit from the power conditioner "fires" the igniter

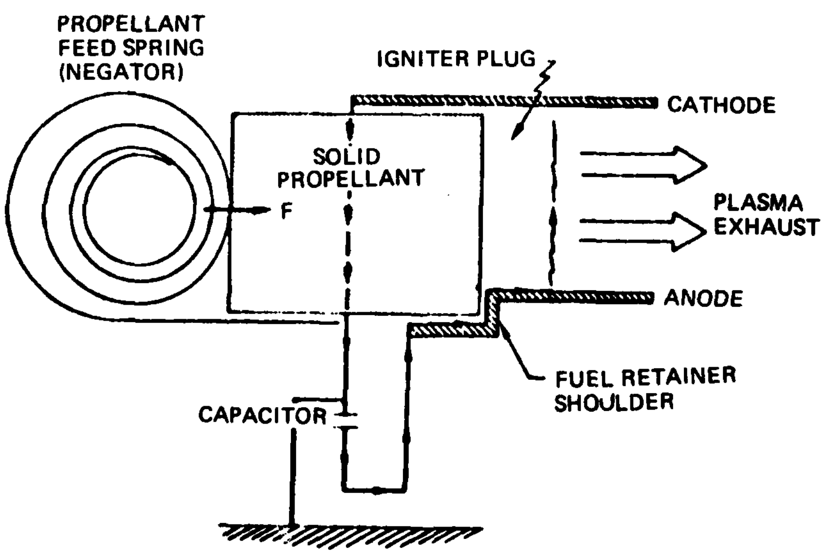

Figure 1. Solid Propellant Pulsed Plasma Thruster Concept (1)

plug. Its micro-discharge produces sufficient electrical conductivity in the thruster nozzle to allow the thruster capacitor to release its stored energy across the face of the Teflon propellant. A few surface layers of propellant become ionized and accelerated by gas dynamic forces and the Lorentz force generated by the interaction between the arc current and its self-generated magnetic field. Since the plasma created is electrically neutral, no charge neutralization is required. After the capacitor has discharged, the cycle can either be repeated or instantly terminated.

In the millipound thruster [3, 4, 5] shown schematically in Figure 2, two helical Teflon fuel bars are fed by negator springs into the discharge chamber. Two fuel retaining shoulders in the anode electrode hold the bars in place. When the discharge is initiated, a plasma bridge is formed which momentarily shorts the electrodes. This completes the circuit for a bank of capacitors totaling 240 microfarads, which are charged to 2500 volts (750 joules). The resultant capacitor discharge sweeps across the Teflon faces and ablates 2.3 μlb of fuel. A plasma slug is ejected by gas dynamic and self-generated electromagnetic forces.

1.4 SYSTEM PERFORMANCE

The solid propellant pulsed plasma millipound propulsion system is designed for large total impulse missions such as north-south stationkeeping. The completely integrated system has recently been vacuum tested for an accumulated 4500 hours. The thruster produces 4.45 millinewtons (1 mlb) of thrust at a specific imulse of 2200 seconds. Other performance parameters are summarized in Table 1.

To provide for the large total impulse, the spring-fed propellant is stored in two helical coils and fed independently into the sides of the thruster's electrode nozzle. The helically coiled propellant rods are suf-

[3] D. J. Palumbo and W. J. Guman, "Pulsed Plasma Propulsion Technology," AFRPL-TR-73-79, September 1973.

[4] D. J. Palumbo, M. Begun, and W. J. Guman, "Pulsed Plasma Propulsion Technology," AFRPL-TR-74-50, July 1974.

[5] D. J. Palumbo and W. J. Guman, "Pulsed Plasma Propulsion Technology," AFRPL-TR-77-40, September 1977.

Table 1. Performance Summary

Impulse bit	5 mlb-sec/pulse
Impulse bit repeatability	±5%, no degradation with time
Thrust (equivalent steady state)	Nominal - 1 mlb Maximum - 3 mlb
Design life (total impulse)	70,000 lb-sec
Specific impulse	2200 sec
System input power	170 W at 1 mlb (1 pulse/5 sec)
Efficiency	
Power processor	0.80
Thruster	0.35
Total System	0.28
Thrust vector accuracy	±0.5 deg
Thrust vectoring	Gimballing if required

ficient for 70,000 lb-sec of total impulse. The thruster electronics subsystem can be either located within the interior of the propellant loop or relocated to the spacecraft equipment compartment to simplify thermal control.

The total volume of the system is about 5430 in^3. The complete system, including propellant, weighs about 100 pounds (45.4 kg).

1.5 MAJOR CONSIDERATIONS IN USING PULSED PLASMA THRUSTERS

The pulsed plasma thruster is a low thrust device, and consequently takes more time to perform thrusting maneuvers than conventional thrusters. Accordingly, thrusting is done for several hours at a time for stationkeeping, and for days in station changing maneuvers. The time for performing a maneuver must be kept in mind when using pulsed plasma thrusters.

The pulsed plasma thruster also requires electrical power to produce reactive thrust. Thus, the spacecraft power subsystem has to be checked to see that it has adequate capacity to power the thrusters and payload. Typical three-axis stabilized communications satellites, as shown in Section 3.5, can accommodate propulsion subsystems using pulsed plasma thrusters within existing power subsystem designs and without additional power penalties.

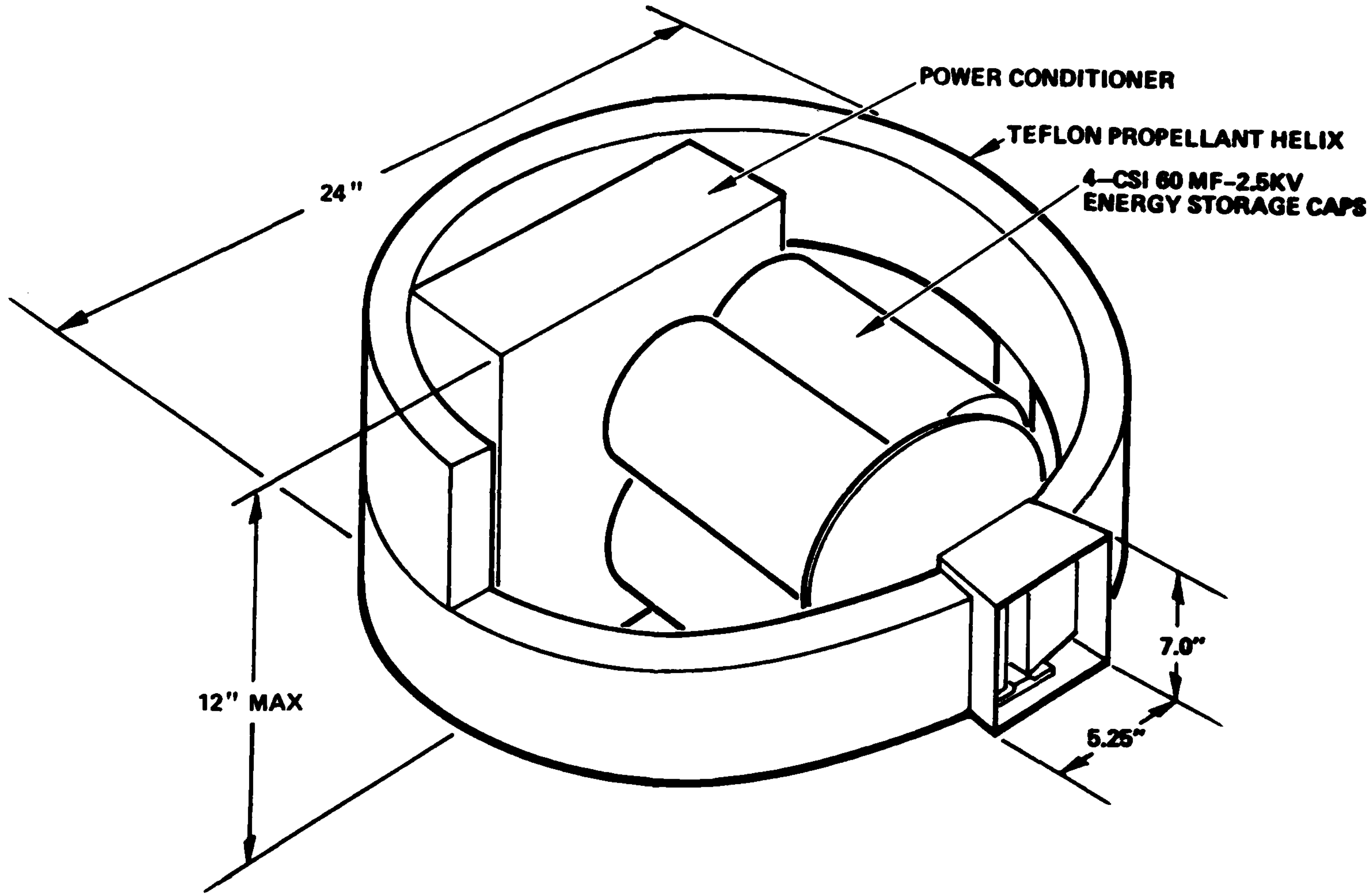

Figure 2. Thruster Schematic Diagram (Internal Components)

Thruster efflux compatibility must be considered in placing thrusters on the spacecraft with respect to locations of nearby spacecraft surfaces. Surfaces immersed in the primary thrust core will be subject to sputtering erosion, whereas surfaces at wider divergence angles from the thrust centerline may be subject to material buildup unless suitably shielded from the exhaust. Section 4.1 describes the thruster exhaust characteristics, and recommends integration approaches for minimizing these effects.

The thruster is an active source of electromagnetic energy, and consequently requires careful attention to electromagnetic compatibility with its host spacecraft. Section 4.2 describes the approach that should be taken to assure successful integration.

1.6 HOW TO USE THIS MANUAL

The manual is organized to present application requirements, equipment descriptions, interface information, and technical data in separate sections. In order to show the interrelationships between sections, a special portion of the manual, Section 2.2, has been prepared to describe the methods involved in selecting a pulsed plasma propulsion subsystem. Section 2.3 shows how to compare the subsystem selected with alternate propulsion means. Thus, the propulsion subsystem designer can go directly to Section 2.2 in order to quickly locate the principal information he needs. It is better, however, to read the entire manual in order to become acquainted with the design data it contains.

Section 2.1 identifies goesynchronous mission requirements that can be performed using pulsed plasma propulsion. The requirements are given for north-south stationkeeping, east-west stationkeeping, station changing, attitude control, and momentum wheel dumping. The method for propulsion subsystem selection is presented, followed by the methods for comparison with alternate propulsive means.

Detailed mechanical, electrical, and thermal interface data are contained in Section 3. Mechanical data include equipment dimensions, weight, mounting provisions, and alignment requirements. Electrical data include input voltage and power requirements, isolation and grounding constraints, input command and output telemetry requirements, and identification of electrical connections on the hardware. Thermal data identify equipment tem-

perature limits and heat dissipation from the equipment when operating. Section 3 concludes with a discussion of three specific mission applications of pulsed plasma propulsion; Defense Satellite Communications System III (DSCS-III), Defense Support Program (DSP), and the Global Positioning System (GPS).

Section 4 discusses the efflux characteristics and resulting design constraints that exist for the use of pulsed plasma thrusters. Electromagnetic compatibility and interactions with satellite communications are also discussed.

Section 5 contains reliability data and recommendations for incorporating equipment redundancy to improve subsystem reliability when needed. Space flight experience is also discussed.

2. SYSTEM ANALYSIS AND SELECTION

Secondary propulsion subsystem mission requirements that can be performed using pulsed plasma propulsion are discussed in detail in this section. These include stationkeeping, station changing, attitude control, momentum wheel dumping, and drag compensation.

The methods for propulsion subsystem selection are then identified, followed by the methods for comparison with alternate means of propulsion.

2.1 MISSION REQUIREMENTS

Typical earth orbiting missions include communications, navigation, meteorology, and earth surveillance satellites. Each specific mission has its own unique requirements. The major functions which may be provided by pulsed plasma propulsion include:

- Spacecraft orbit control (north-south, east-west stationkeeping and station changing)
- Attitude control (active control or momentum dumping)
- Maintenance of orbital velocity

Initial station acquisition involves removing the launch vehicle injection errors so as to attain the desired orbital flight path. In addition, it generally requires moving the satellite from the injected in-track position to the desired in-track location. When the Shuttle Transportation System (STS) is fully operational, initial station acquisition will be provided by STS, thus eliminating the need for initial acquisition maneuvers. Subsequent station changing maneuvers are functionally identical to the initial station acquisition. Higher rates of change, however, may be required.

Orbital control and drag compensation functions, with their attendant large propellant expulsion requirements, provide the most natural applications for pulsed plasma propulsion. This is especially true for north-south stationkeeping. Attitude control is generally applied to symmetric spacecraft configurations which minimize disturbance torques, hence propellant consumption. The low impulse bits attainable with pulsed plasma propulsion make it well suited for attitude control and precision velocity maintenance.

Pulsed plasma attitude control, with its inherently low propellant consumption, allows these traditional spacecraft configuration constraints to be relaxed, hence saving structural as well as propellant weight and providing design flexibility. The following sections discuss many of the factors pertaining to these applications.

2.1.1 Stationkeeping

Stationkeeping involves changing the satellite velocity vector to correct for orbital perturbations which are caused primarily by nonsymmetric earth and lunisolar gravitational forces. The perturbations are of a complex oscillatory nature with component periodicities of about 12, 12.5, 24, 25 hours, 1 month, 1, 2, 18, 53 years, etc. Consider that the components having periods equal to the orbital period or less as the short-term effects, and the components having periods greater than the orbital period (i.e., on the order of a month or greater) as the long-term effects, then the short-term accelerations, which cause diurnal variations in a satellite's unperturbed orbit, are on the average, four or more times greater than the long-term or secular accelerations. It is, however, the long-term secular position drifts in-plane and inclination buildup out-of-plane that cause the satellite to inhabit an ever widening volume of space relative to the earth.

In most missions, correcting the secular perturbing forces will provide the needed satellite orbital position accuracy. However, for very precise navigational satellites, particularly when attempts are made to reduce user equipment to a minimum, the diurnal perturbations have to be considered.

2.1.1.1 North-South Stationkeeping

The orbit plane of a synchronous orbit satellite tends to drift largely because of the combined effects of earth oblateness and lunisolar gravitation.(6) The important orbital parameters are the amount of orbital correction required, the direction of orbital correction, the maximum time interval between orbital corrections, and the penalty incurred by not making orbit corrections at the exact time desired.

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[6] G. S. Gedeon and M. T. Palmitter, "Orbit Manual for Synchronous and Subsynchronous Satellites," TRW Report 99900-6310-R000, August 1967.

For small inclinations, the rate of change of inclination as a function of modified Julian date is:

$$di/dt \text{ (deg/year)} = 0.8475 + 0.0985 \cos (0.0533t - 2149)$$

where t is the modified Julian date in years (counted from May 24, 1968) and the cos argument is in degrees. This translates to yearly rates ranging from 0.75 to 0.95 deg/yr or 132 to 167 ft/sec/yr. The exact yearly values are given in Table 2. It is desired to fire stationkeeping pulses at those two points in orbit which allow exact cancellation of the precession of the orbit plane. The direction of orbital precession is nearly

Table 2. North-South Stationkeeping Requirements

Year	Inclination Rate (deg/yr)	Δv (ft/sec)
1975	0.795	139.9
1976	0.770	135.6
1977	0.754	132.7
1978	0.748	131.6
1979	0.753	132.6
1980	0.769	135.3
1981	0.794	139.7
1982	0.824	145.0
1983	0.857	150.9
1984	0.889	156.5
1985	0.916	161.3
1986	0.936	164.7
1987	0.945	166.4
1988	0.944	166.1
1989	0.931	163.9
1990	0.909	160.0
1991	0.880	154.9
1992	0.848	149.2
1993	0.815	143.4
1994	0.786	138.3

inertially fixed and the place in orbit is a function of the time of year. Figure 3 illustrates the phenomenon. The desired orbital times of day to perform stationkeeping are near 6 AM or 6 PM at the spring and autumn equinoxes and near noon or midnight in the winter or summer solstices.

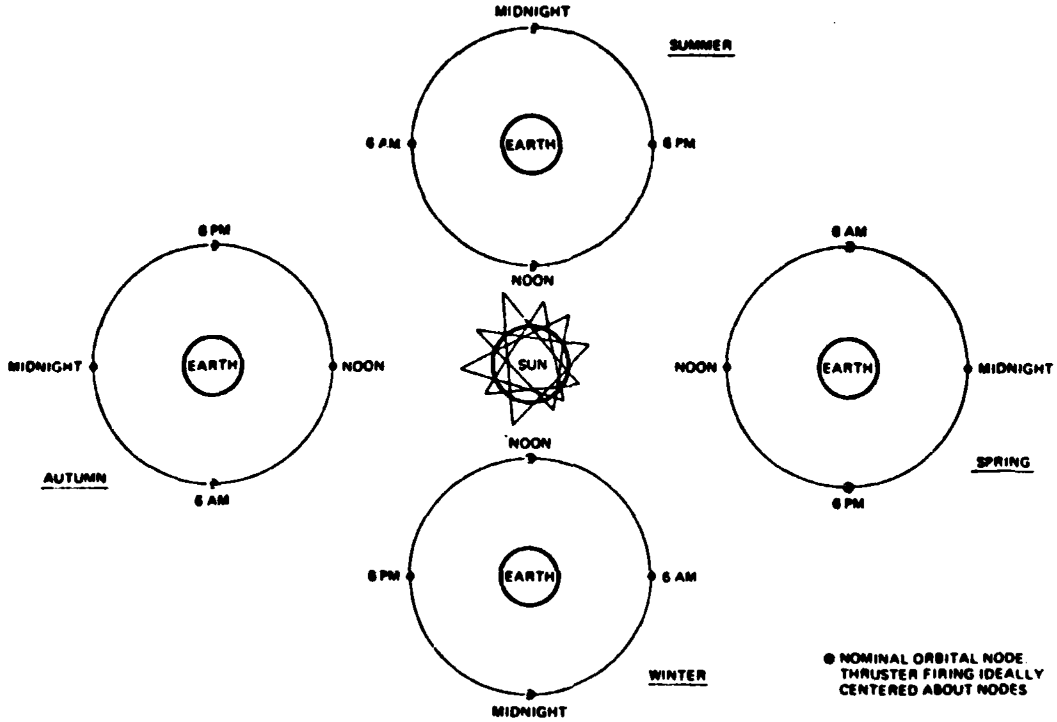

Figure 3. Geometry of Orbit Corrections

Making orbital corrections away from the desired nodes affects thrusting efficiency since the orbit axis is not being precessed exactly opposite to that of drift. This loss of efficiency is roughly proportional to $(1 - \cos \varepsilon)$ where ε is the orbital angle between the node and the point where the thruster was fired. This translates directly into increased propellant consumption.

If the thrusting is distributed over a significant percentage of the orbit (>10 degrees) as would certainly be the case with pulsed plasma

propulsion, efficiency is lost by the ratio sin α/α where α is the orbital half-angle over which thrusting takes place.

Thrusting inefficiency is also introduced when the thrusters are canted away from the north-south axis. When the thrusters are body mounted on three-axis stabilized spacecraft, canting is implemented to minimize effluent interaction with the solar arrays. For a cant angle ϕ with respect to the north-south axis, a cant angle efficiency factor, cos ϕ, is introduced.

Combining the above factors, the required number of thrusting hours per node, t, is given by:

$$t = \frac{24}{\pi} \sin^{-1} \left[\frac{\pi}{(24)(32.2)(3600)} \cdot \frac{1}{N_N N_D} \cdot \frac{M \Delta V}{F \cos \phi} \right] \qquad (1)$$

and the total operating life per thruster is given by:

$$L = N_D N_Y t \text{ hours} \qquad (2)$$

where

M = spacecraft mass, lb (assumed to be constant, i.e., neglecting cumulative propellant expulsion as mission progresses)

ΔV = yearly mission velocity increment, ft/sec/yr

F = thrust, pounds

N_N = number of nodal firings/day

N_D = number of days firing/year

N_Y = number of years/mission

These effects are shown graphically in Figure 4, where orbital efficiency and thrust are plotted versus thrusting time at each node for a typical 2000-pound spacecraft. Power requirements are also shown for pulsed plasma propulsion at 170 watts/millipound. North-south stationkeeping is performed daily to provide ΔV of 150 ft/sec/yr. Thrusting time requirements are shown for cant angles of 0 and 30 degrees. If stationkeeping is done

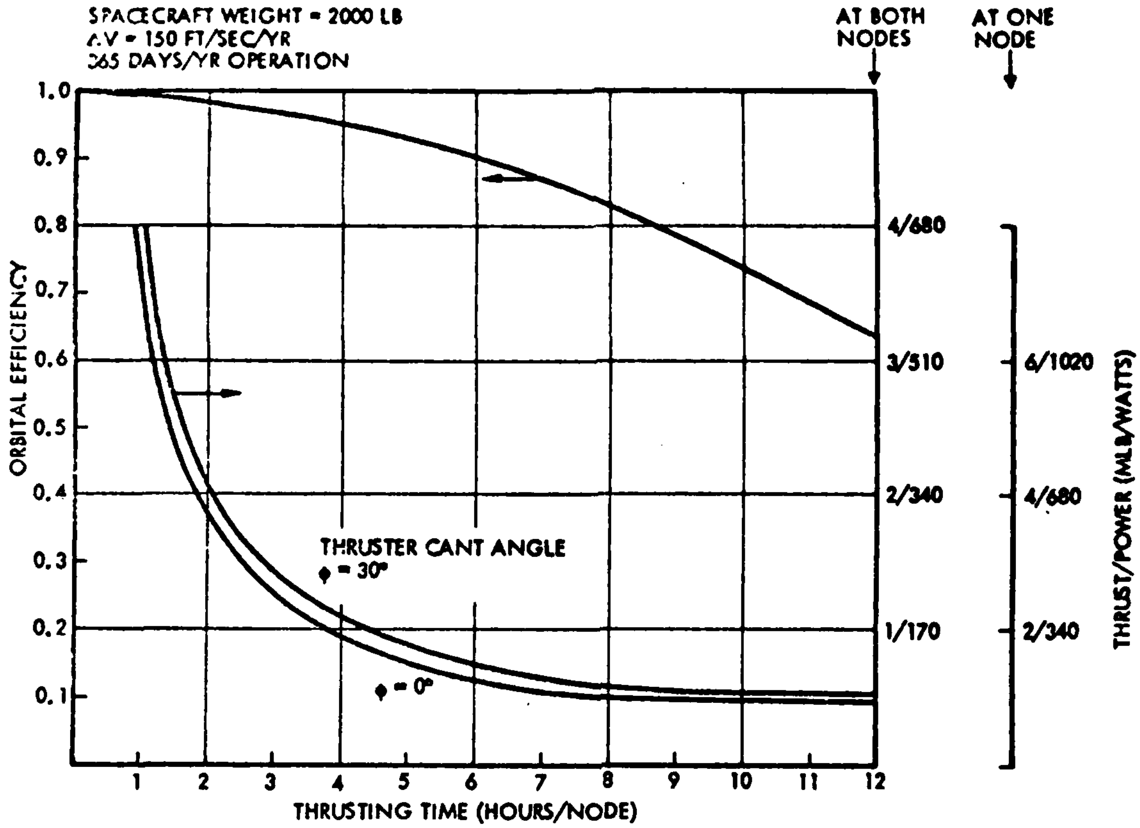

Figure 4. Orbital Efficiency and Thrusting Time for North-South Stationkeeping

at both nodes with a pair of millipound thrusters canted 30 degrees, then each thruster operates for about 2 hours a day.

When the north-south stationkeeping thrusters are canted into the orbit, the in-plane component of the impulse affects the satellite orbit path. A component along the velocity vector changes the orbital period and eccentricity. A component normal to the velocity vector changes the orbital eccentricity. The effects on the orbit because of this cant have been studied via a digital simulation of the orbit.[7] If the induced eccentricities are uncompensated by firing at the opposite node, they will continue to build up

[7] "Electric Propulsion/Spacecraft Integration Study - Final Report," TRW Report 26951-6001-TU-00, Comsat Contract IS-680, March 1976.

to an unacceptable level. The most straightforward form of compensation is to fire a complementary pulse (with respect to north or south thrusting) 12 hours after the initial firing. East-west stationkeeping or compensating canting of one thruster is employed to eliminate the residual drift.

2.1.1.2 East-West Stationkeeping

The triaxiality of the earth's mass distribution exerts a small force east or west along the satellite velocity vector toward one of the gravitational equilibrium points. Figure 5 summarizes the annual drift as a function of spacecraft longitude. For example, for a station at 240°E, the annual velocity drift is 2 ft/sec.

The nominal stationkeeping interval in days can be approximately described by

$$P = c(\lambda)\ \sqrt{DZ} \quad (3)$$

where

P = stationkeeping interval, days

c(λ) = a constant, depending on longitude

DZ = stationkeeping dead zone, degrees

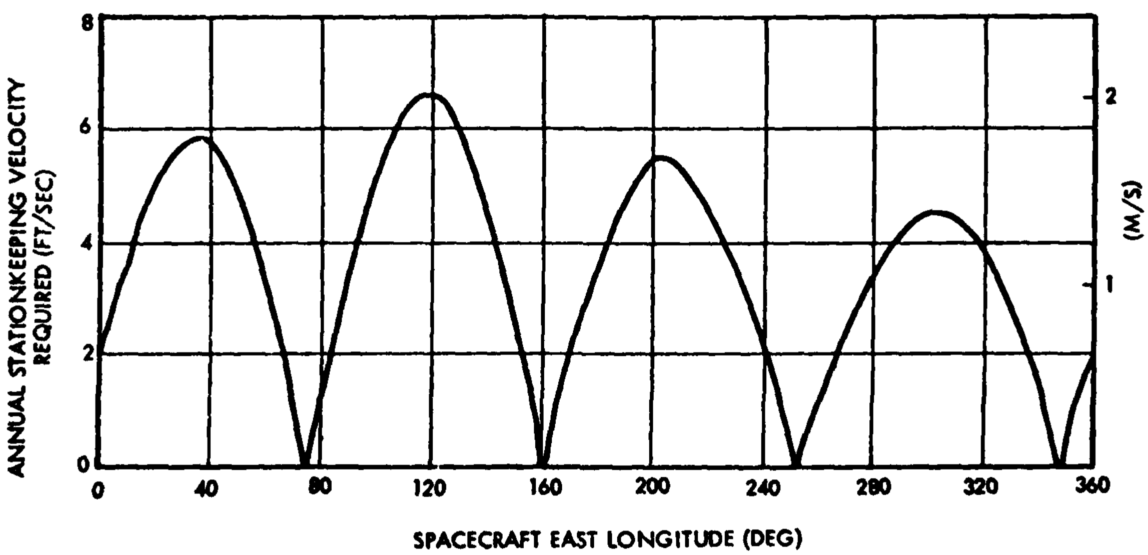

Figure 5. Annual Longitude Stationkeeping Velocity

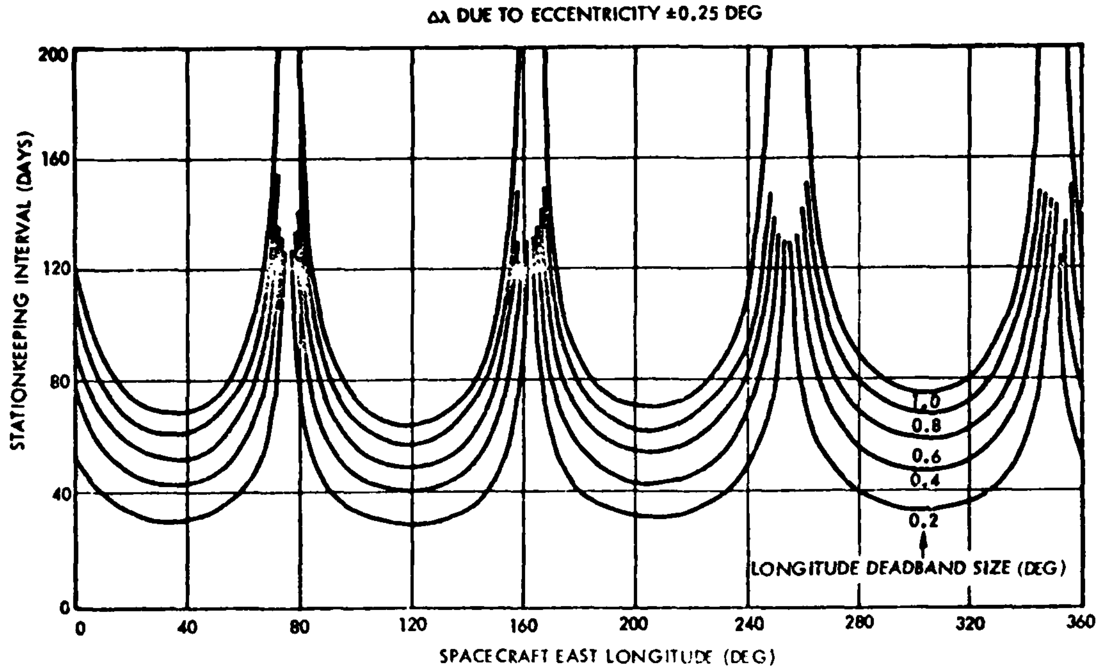

Figure 6. Limit Cycle Time for Stationkeeping

Figure 6 illustrates this relationship graphically. If, for example, it was desired to maintain east-west stationkeeping within ± 0.1 degree at 240°E longitude, the procedure below would be followed. From Figure 6 a value for c (240°E) can be obtained by reading P = 50 days for DZ = 0.2 degree and then using Equation (3) to solve for c (240°E). Finally, P for a dead zone of 0.1 may be obtained from Equation (3)

$$P = \frac{50}{\sqrt{0.2}} \sqrt{0.1}$$

$$P = 35 \text{ days}$$

With this Δv interval, then, the Δv/cycle is

$$\Delta v/\text{cycle} = \frac{35}{365.24} (2) = 0.22 \text{ ft/sec/cycle}$$

Thus, east-west stationkeeping for ± 0.1 degree: Δv = 0.22 ft/sec every 35 days.

2.1.2 Station Changing

Orbital station changes are required when the desired earth surveillance area changes or the spacecraft replaces a failed spacecraft.

The repositioning velocity increment for a synchronous orbit is:[8]

$$\Delta v = 18.7 \frac{\theta}{t_\theta} \tag{4}$$

where

Δv = velocity increment, ft/sec needed to reposition for an impulsive maneuver

θ = repositioning angle, degrees

t_θ = time for repositioning maneuver, days

Figure 7 compares impulsive and continuous thrusting maneuvers. In the figure, the areas under the curves, velocity times time = distance must be

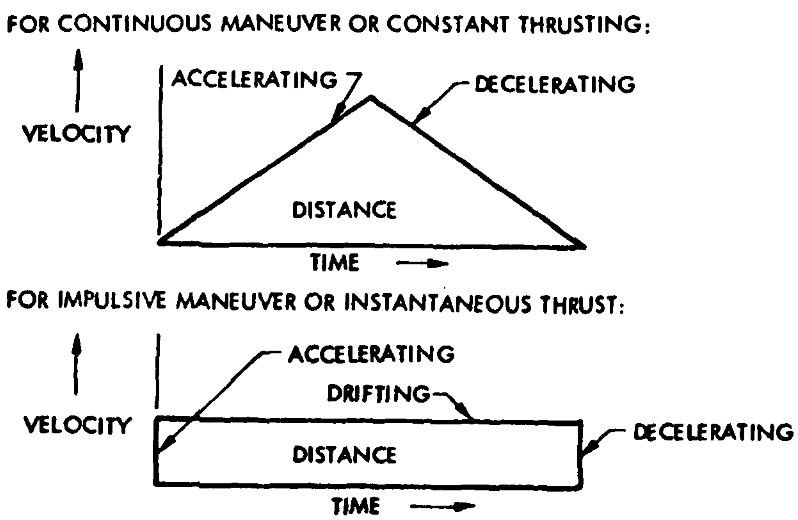

Figure 7. Comparison of Continuous and Impulsive Station Changing

[8]G. S. Stern, "A Brief Description of the Synchronous Satellite Mission," TRW IOC 3431.5-240, 2 November 1968.

the same because the distance to be traveled, θ^o is the same. If the areas are equal, then the height of the triangle must be twice the height of the rectangle. Therefore continuous thrusting requires twice the final velocity (or starting from 0 velocity, twice the Δv) that is required for impulsive thrusting during a repositioning maneuver.

The allowable time to move a satellite from one position in an orbit to another position in the orbit influences the selection of the thrusters to perform this function. Figure 8 illustrates the minimum continuous thrust levels required to move a 1000-pound satellite to a new location in a synchronous orbit during a specific time interval. Since the required thrust level is directly proportional to satellite weight, minimum thrust levels required for other satellites can easily be obtained.

The minimum required thrust is inversely proportional to the square of the maneuver time. As Figure 8 shows, to move 10 degrees in 10 days requires a few millipounds of thrust; to move 10 degrees in 1 day increases the thrust

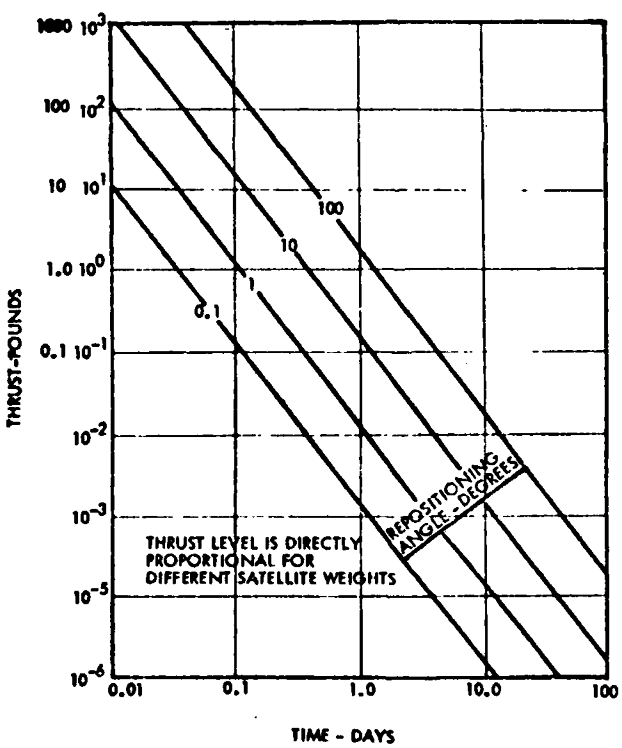

Figure 8. Minimum (continuous) Thrust Level Required to Reposition a 1000-Pound Geosynchronous Satellite

to over 0.1 pound. From the figure, the maneuver time and total thruster operational time are equal; i.e., thrusting is continuous.

2.1.3 Attitude Control Functions

Geosynchronous spacecraft of the 1985 to 1990 time frame have attitude control pointing requirements that vary with mission requirements. Communications satellites have pointing requirements in the range of 0.03 to 0.3 degree accuracy. Surveillance satellites require extremely precise attitude determination capability (on the order of arcseconds) and jitter (on the order of 10^{-5} deg/sec) with absolute pointing accuracy of 0.01 to 0.05 degree. Specific requirements depend on ground interface equipment and satellite payload sensors. These accuracies can be met with high momentum spin stabilized satellites or three-axis stable satellites utilizing reaction or momentum wheels for intermediate momentum storage and thrusters or magnetics to unload the wheels. The three types of attitude control that appear most likely are: (1) a spinner with periodic precession of the momentum vector by propulsion, (2) the simple momentum-biased three-axis control system such as used in the new generation of communications satellites (FLTSATCOM, CTS, OTS, RCA-GlobeCom, Intelsat V and TDRS), and (3) a four-reaction wheel,* three-axis control system which provides the most accurate pointing.

The pulsed plasma stationkeeping subsystem can be used directly to perform these attitude control functions or to unload momentum. Section 3.5 gives examples of how this is done. Attitude control propellant requirements are small compared to stationkeeping requirements, typically less than 10% of that required for north-south stationkeeping.

Low earth orbiting spacecraft generally use momentum transfer techniques for attitude control, rather than mass expulsion. A pulsed plasma drag compensation subsystem may be gimbal mounted to provide for momentum wheel unloading, but this may not be desirable because magnetic torquers are very effective at low altitudes. (At geosynchronous altitude, the earth's magnetic field is so weak that magnetic torquers lose their effectiveness.)

*Three wheels are required for three-axis control. The fourth wheel is for redundancy.

2.1.3.1 Thrust Vectoring

The pulsed plasma thruster may be gimbal mounted to provide a thrust vectoring capability. Vectoring is desirable to minimize disturbance torques on body-stabilized spacecraft and to maintain spin-rate on spin-stabilized spacecraft. It is also used to allow a single thruster to fulfill multiple functions.

A practical thrust vectoring system for long duration missions must simultaneously satisfy a number of stringent requirements:

- Adequate thrust vector range
- Adequate thrust vector resolution
- Minimal degradation of other thruster functions
- Sufficient mechanical integrity throughout launch environment

The required vectoring range depends on the total spacecraft requirements and specific mission responsibilities (e.g., attitude control and stationkeeping). If the thruster is used for stationkeeping only, the thrust vector range requirements are determined by the need to reduce thrust misalignment torques to a level compatible with the attitude control system. In this case, the required vectoring range is relatively small since the unperturbed thrust vector can be located within 1 degree. However, high angular resolution within the range is desirable.

Thrust vector resolution is potentially important as it directly affects the attitude control subsystem. For a 7- to 10-year mission, a small misalignment can result in a high total impulse requirement for the attitude control system. For this reason, it is desirable to fine tune the vectoring angle as accurately as possible. The ability to resolve the vector angle to tenths of a degree is therefore desirable.

Another attractive feature of thrust vectoring is that it allows thrusters to be positioned on the tips of the solar arrays without requiring the solar array axis to pass through the mass center. The main advantage of placing the thrusters on the array tips is for enhanced effluent compatibility. However, this benefit is traded for a more complex interface problem (since the arrays rotate) and the added weight required to stiffen the array booms.

Because of the rotation of the arrays, frequent revectoring is required to maintain the thrust vector pointing through the spacecraft center of mass. Body-mounted thrusters do not have these requirements.

Thrust vectoring of pulsed plasma thrusters is also useful for attitude control purposes. For instance, if the vectoring loop is closed through ground software, the disturbance torques may be used to control spacecraft attitude or unload reaction wheel momentum. This may also be accomplished open loop. Thus, thrust vectoring permits a stationkeeping thruster to simulatneously perform attitude control functions.

2.1.3.2 Momentum Wheel Dumping

In order to achieve the required pointing accuracies of the new generation of attitude control systems, interim momentum storage via reaction wheels or momentum wheels is required. This permits fine tuned attitude control and provides a method for handling periodic disturbance torques without the need for mass expulsion.

A momentum wheel concept uses a momentum bias whereas a reaction wheel system does not. A single momentum wheel can provide control about three axes by its gyroscopic action, whereas a single zero momentum reaction wheel can provide torque about only one axis. In the momentum bias approach, one or more continuously running wheels are used to provide a net momentum vector perpendicular to the orbit plane. Gyroscopic stiffness is achieved in both roll and yaw axes and reduces attitude error buildup caused by disturbance torques. Rotation about the pitch axis is controlled by variations in wheel speed. The spacecraft rotates at orbit rate so that there is a kinematic coupling between roll and yaw errors. The momentum bias stiffness maintains attitude roll/yaw errors within required limits until the inertial error can be measured with a roll sensor and corrected. This approach obviates the need for a yaw sensor, except for high torque time periods caused by stationkeeping thruster misalignments. With thrust vectoring of the 1-mlb thrusters, however, disturbance torque levels are on the order of magnitude of solar pressure disturbances. Hence, for a momentum-bias system, and adequate confidence in the predictability of the thrust vector, yaw sensing is not required during stationkeeping. Increasing the wheel size may be required.

The three-axis zero momentum approach uses three or more wheels. The wheels generate reaction torques to counter disturbance torque perturbations. In this approach a yaw sensor, as well as a pitch and roll sensor, is required. Zero bias systems have the potential for higher yaw accuracy and can perform with larger disturbance torques independent of the momentum storage implementation selected. Unloading of these wheels due to long-term steady-state disturbances (or short-term transients which exceed wheel capacity) is required. The usual approach is via mass expulsion (thrusters). This can be done by a secondary propulsion subsystem designed to provide torques only or an orbit control propulsion subsystem designed to provide the desired torque and a linear acceleration simultaneously. This occurs with the pulsed plasma subsystem if wheel unloading is performed during a stationkeeping maneuver and the thruster is vectored to provide a desired torque.

The momentum storage devices are generally sized to minimize weight but store the momentum for at least 1 day (since many periodic torques are of a 24-hour period). In order to minimize weight, it is desired to unload the wheels at least every 1 or 2 days. A pulsed plasma thruster stationkeeping schedule consistent with this schedule can be used to unload the wheels with essentially no increase in propellant consumption (since this is required for stationkeeping anyway).

The principal synchronous orbit steady-state distrubance torques that cause the wheel momentum to grow are caused by solar disturbance torques. Because of the propellant requirements of conventional hydrazine, present spacecraft are designed to minimize solar torques. This requires a spacecraft symmetry with severe restrictions on placement of large antennae and generally dictates two similar solar array panels with separate solar array drive mechanisms. Since pulsed plasma propulsion can provide substantial impulses with minimal weight penalty, a spacecraft with pulsed plasma attitude control can be designed with large asymmetries--for example, a single solar array panel. This benefits pulsed plasma propulsion since the thrusters can be body-mounted on the free side of the spacecraft, and efflux compatibility concerns minimized. To provide quantitative examples, typical current generation communications satellites with symmetric solar panel designs have constant solar disturbance torques ~5 x 10^{-6} ft/lb which trans-

lates to a 10-year impulse of 1500 ft/lb/s. For single solar array panel design, this torque may increase a factor of 20 to 10^{-4} ft/lb in unloading torque required. By contrast, the single solar array panel design saves weight in the solar panel drive mechanism and may be required to provide the geometry needed to accommodate very large, body-mounted payload components.

2.1.3.3 Yaw Sensing for Systems with Three-Axis Control

A three-axis stable satellite controls pointing simultaneously about the pitch, roll, and yaw axes. Yaw control for momentum-biased systems is achieved passively due to the kinematic coupling of the roll and yaw axes. This type of control is possible only when the disturbance torques are small, such as they normally are for synchronous orbit satellites. Active yaw control is required when the disturbance torques are large. During the active yaw control phase, a yaw sensor is required.

Sun sensors represent the simplest devices that may be employed to obtain yaw information. They are space-proven, highly reliable, of long life, and weight effective. The main disadvantage in the use of sun sensors for this application is that they cannot provide a yaw reference when the satellite, earth, and sun are collinear. This occurs near the solstices and can overlap the thrusting period for as many as 120 days per year. One solution to this problem is to perform an in-flight thrust vector alignment and then demonstrate subsequent thrust vector reproducibility to better than a few tenths of a degree. Another solution is to use a gyro in place of the sun sensor. Alternately, there are 245 days per year when a good reference is available which allows adequate time to maintain the orbit inclination to within a ± 0.1 degree drift requirement. However, since thrusting cannot be performed every day, thrusting must be performed at a greater distance from the orbit nodes. Thus thrusting is less efficient and there are resultant propellant and operational life penalites as discussed in Section 2.1.1.

2.1.4 Drag Makeup Velocity Increment Determination

Low altitude missions have, in general, spanned a rather narrow band of altitudes. The orbital altitudes are constrained by drag limitations and their effect on orbital lifetime for the lower altitudes and by radiation hazards or the sensor resolution limitations for the higher ones. Once the

orbit altitude and inclination have been selected, perturbing forces must be contended with in order to retain the selected flight path.

The major influence on the lower orbits is the density of the atmosphere and the resultant drag it imparts to the satellite. Figure 9 illustrates the aerodynamic drag per frontal area as a function of altitude for a typical orbit velocity.

The lowest altitude at which pulsed plasma propulsion would be attractive would be one for which the thrust significantly exceeds the additional drag due to the added solar array. If an array power factor of $\sim$ 15 W/ft^2 is assumed, one 170-watt thruster system requires 11.3 ft^2 of additional array. A drag of 1 millipound due to this array corresponds to a drag factor of 8.85×10^{-5} lb/ft^2 (4.23×10^{-3} N/m^2). From Figure 9, this corresponds to an altitude of 275 km. Thus, the millipound pulsed plasma propulsion system should only be considered for drag makeup for altitudes in excess of 275 km.

2.2 SUBSYSTEM SELECTION

The steps followed in designing a secondary propulsion subsystem using pulsed plasma thrusters are:

(1) Calculating the thrust level

(2) Calculating propellant requirements

(3) Selecting equipment

(4) Identifying power and weight allocations

(5) Selecting thruster locations on the spacecraft

(6) Trading off thrust level/system weight/reliability

Frequently, propulsion subsystem design is an iterative process as design constraints are encountered in proceeding from steps (1) through (6) which require trade comparisions to be made for particular applications in order to arrive at the optimum system design. Qualitatively, power and operating time requirements are nearly proportional to thrust level. Propellant weight requirements are nearly fixed by mission parameters, while propulsion subsystem dry weight is dependent on reliability and redundancy considerations.

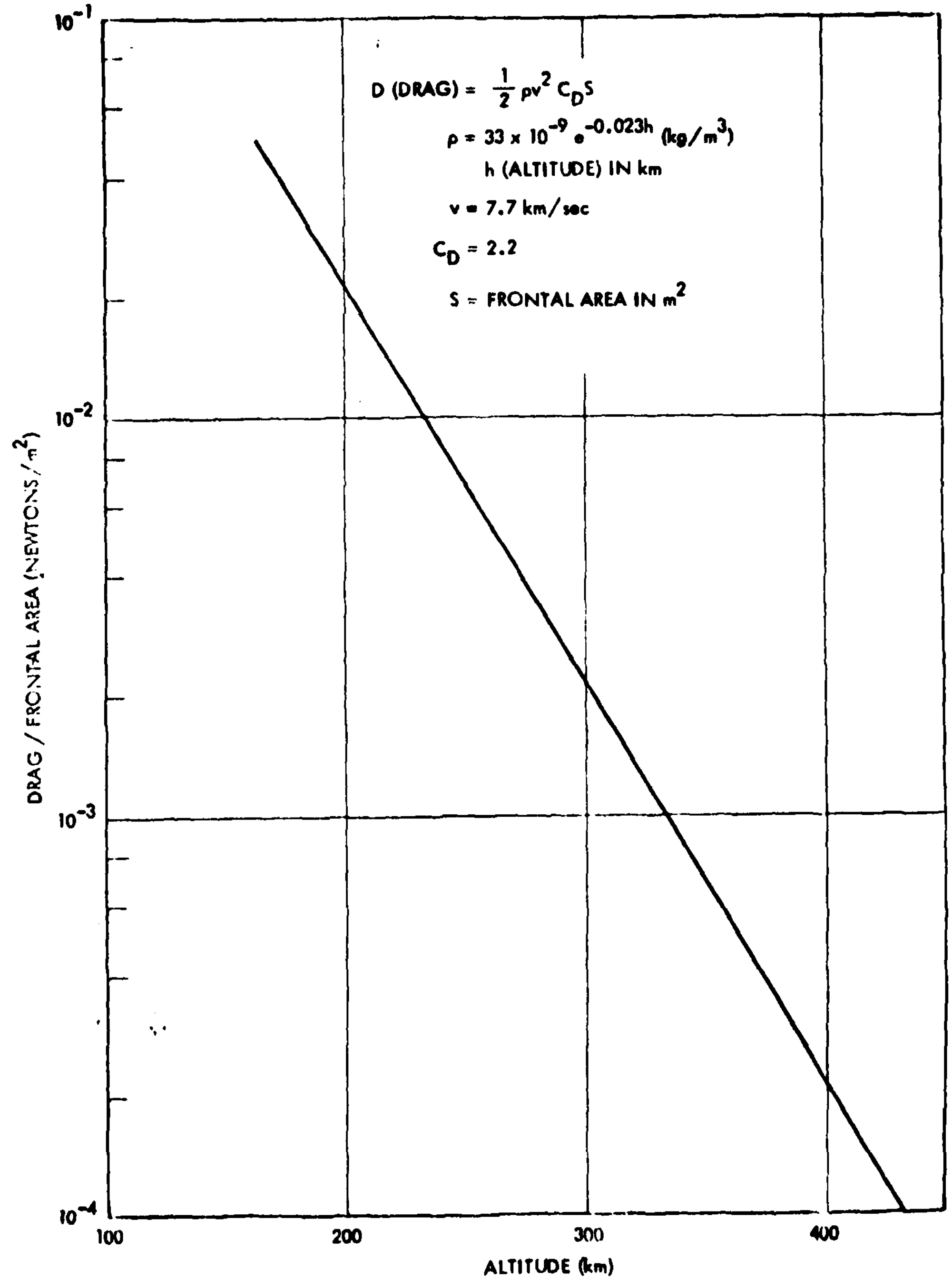

Figure 9. Aerodynamic Drag per Frontal Area as a Function of Altitude (mean CIRA atmosphere) (Spacecraft Orbit Velocity of 7.7 km/s)

2.2.1 Step 1: Calculating the Thrust Level

For large ΔV maneuvers, the thrust level is closely related to thruster operating time because these maneuvers are performed over relatively long periods of time with the low average thrust provided by pulsed plasma thrusters. Attitude control requirements may be satisfied by gimbal mounting the thrusters and vectoring them during ΔV maneuvers. Alternatively, pulse trains of 5 mlb/s/pulse, at repetition rates up to 1 pulse every 5 seconds, may be used for active attitude control. Thus the average thrust level is given by:

$$F = 5 \times 10^{-3}\ nf \qquad (5)$$

where

F = average thrust, millipounds

n = number of thrusters

f = pulse frequency, $(\text{sec})^{-1}$

Each thruster has a design life of 1.4×10^7 pulses (thereby yielding a maximum total impulse capability of 70,000 lb-sec). When selecting thrust level, it is desirable to minimize the number of pulses, i.e., the total operating time, per thruster in order to keep overall system reliability high.

The thrust level for north-south stationkeeping is determined with the aid of Equation (1). A typical design curve, shown in Figure 4, was prepared for a 2000-pound spacecraft with daily stationkeeping. For this case, with a millipound thruster operating at each node (two thrusters total, one on the north side and one on the south side of the spacecraft and thrusting through the center of mass), the thrusters operate for 3.7 hours at each node. By keeping the thrusting time per node small, orbital efficiency is improved.

The total operating time per thruster for north-south stationkeeping is obtained from Equation (2).

The thrust level and time for station changing maneuvers are determined from Figure 8, which was prepared for a 1000-pound satellite. Since the required thrust level is directly proportional to satellite weight, thrust levels required for other satellites are easily obtained.

East-west stationkeeping requirements are small compared to north-south stationkeeping or station changing, and like attitude control requirements, may be implemented by vectoring the north-south thrusters or using the station changing thrusters directly.

The thrust level for drag makeup is derived from the equation:

$$F = C_d\ (A_s + 11{,}300F) \qquad (6)$$

F = thrust (lb)

A_s = satellite cross section (ft^2)

C_d = drag/frontal area (lbf/ft^2)

where the second term in Equation (6) is the drag due to the additional array required for the thruster, thus:

$$F = \frac{C_d\ A_s}{1 - 11{,}300\ C_d} \qquad (7)$$

As an example, at 300 km $C_d = 2 \times 10^{-3}\ N/m^2$ ($4 \times 10^{-5}\ lb/ft^2$), and for a 50 ft^2 satellite:

$$F = \frac{4 \times 10^{-5}\ (50)}{1 - 0.452} = 3.65 \times 10^{-3}\ lb$$

2.2.2 Step 2: Calculating Propellant Requirements

Propellant requirements for the pulsed plasma thruster are calculated from

$$m_p = 2.27 \times 10^{-6} n_p \qquad (8)$$

or $$m_p = 4.54 \times 10^{-4} \times \text{total impulse (lb-sec)} \qquad (9)$$

where m_p = propellant mass, lb

n_p = number of pulses

For estimating purposes, thruster propellant requirements may be calculated from

$$m_p = \frac{M\,\Delta V}{(32.2)(2200)} \text{ pounds} \tag{10}$$

where M is the spacecraft mass in pounds, ΔV is the required velocity increment in ft/sec, 32.2 ft/sec^2 is the gravitational constant, and 2200 seconds is the rated specific impulse of the pulsed plasma thruster.

The total thrusting time for each north-south stationkeeping thruster is given by Equation (2). The total station changing time is obtained from Figure 8. If east-west stationkeeping and attitude control can be performed simultaneously with these larger ΔV maneuvers by taking advantage of the thruster gimbal mounting, then no additional propellant is required. If not, additional propellant needs may be estimated from Equation (10) or by setting aside an additional 10% of the north-south stationkeeping propellant mass for attitude control.

For drag compensation the propellant requirement is calculated simply from the product of the average thrust level (F) times the mission duration (T). Thus:

$$m_p = \frac{F\,T}{(32.2)\ 2200} \text{ pounds} \tag{11}$$

2.2.3 Step 3: Selecting Equipment

The performance characteristics of a 1-millipound pulsed plasma thruster are listed in Table 3. Each thruster consists of two major assemblies: (1) the propellant-discharge assembly, and (2) the power conditioner. These two assemblies may be packaged together in a single unit as shown in Figures 2 and 10, or they can be physically separated on the spacecraft. Figure 2 is an isometric representation of parts within the single package; Figure 10 is a photograph of the development model.

Table 3. Thruster Performance Summary

Impusle Bit	5 mlb-sec/pulse
Impulse Bit Repeatability	± 5%
Equivalent Steady-State Thrust at 1 Pulse/5 sec*	1 mlb
Input Power at 1 Pulse/5 sec*	170 watts
Specific Impulse	2200 sec
Mass Flow	2.27 μlb/pulse
Efficiency	
Propellant Discharge Assembly	35%
Power Conditioner	80%
Total	28%

*Equivalent steady-state thrust level and average power vary directly with pulse frequency.

Once the thrust level and propellant requirements for the mission have been determined, the number of pulsed plasma thrusters may be selected in preparing the components list for the propulsion subsystem. Each propellant-discharge assembly is rated for 3 mlb (maximum) equivalent steady-state thrust and 1.4 x 10^7 (maximum) pulses at 3 pulses/5 seconds. The power conditioner is rated for 1 mlb (maximum) equivalent steady-state thrust. The power conditioner, therefore, limits the maximum available thrust level whereas the propellant-discharge assembly limits the maximum available total impulse form each unit. Propellant may be off-loaded in each assembly employed, i.e., the maximum allowable propellant loading need not be carried at each thruster location.

2.2.4 Step 4: Identifying Power and Weight Allocations

Section 3 contains detailed interface information for equipment used in pulsed plasma propulsion, including hardware weights and power requirements.

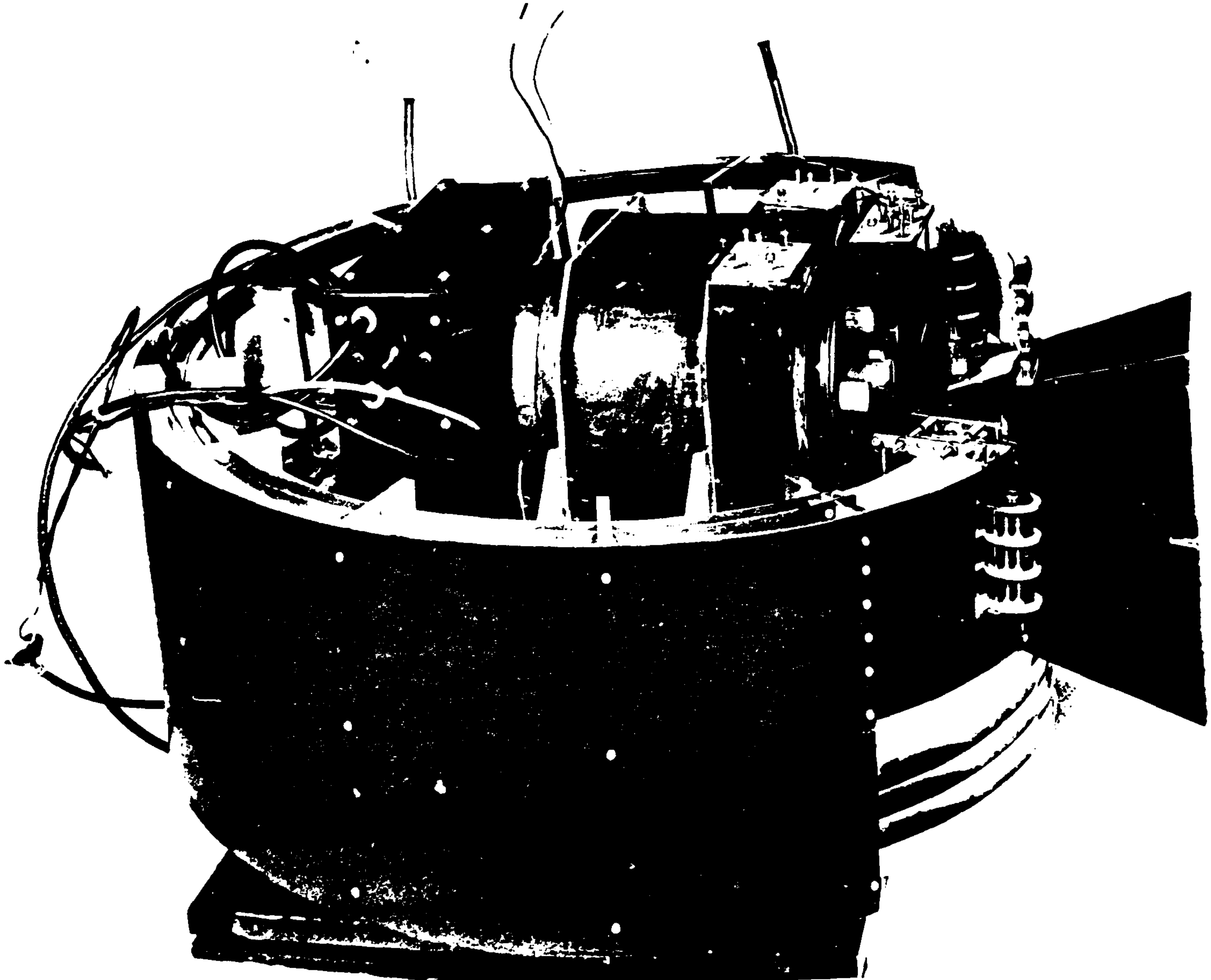

Figure 10. Development Model, One-Millipound Pulsed Plasma Thruster

From the thrusting schedule that was determined in Step 1 for performing propulsive functions, and from the interface data, a propulsion system power profile is readily prepared.

The propulsion subsystem weight allocation is obtained by adding up the weights of all the equipment and interconnecting cabling that comprise the subsystem. Depending on mass properties allocation methods for particular spacecraft, structural weight may have to be added for special mounting brackets, supports, or cable ties. In most cases, these considerations have already been included in the structural mass allocation for the spacecraft. The spacecraft power subsystem weight has to be reviewed to assure that it can accommodate the propulsion subsystem requirements within its existing mass allocation. Otherwise, a power subsystem weight increase for propulsion has to be assessed either against propulsion or power, depending on bookkeeping methods. The example given in Section 3.5 for a 2000-pound geosynchronous communications satellite shows how pulsed plasma propulsion was incorporated within the existing power subsystem design and without any weight increase specifically needed for propulsive power requirements.

2.2.5 Step 5: Selecting Thruster Locations on the Spacecraft

Nominal thruster orientations on the spacecraft were determined in Step 1 from the required vector locations. Detailed mounting and orientation considerations, however, are very dependent on particular spacecraft configurations. Interactive effects between pulsed plasma thrusters and host spacecraft are discussed in Section 4. The data from that section are used in determining the ultimate thruster locations with respect to other spacecraft surfaces, with particular attention given to plume effects and electromagnetic compatibility.

2.2.6 Step 6: Trading-Off Thrust Level/System Weight/Reliability

The propulsion subsystem reliability is calculated using the failure rate data presented in Section 5. Reliability may be improved by reducing thruster operating time, i.e., increasing the thrust level for a given propulsive function, or by adding redundant components. Both of these approaches add dry weight to the propulsion subsystem. Thus, a tradeoff exists between thrust level, weight, and reliability for many applications.

As mentioned earlier, propulsion subsystem design is an iterative process and the steps outlined have to be repeated to investigate alternative approaches before the final design is chosen. Nonetheless, an estimate of the final configuration characteristics can be rapidly made by following these steps before proceeding through all the tradeoff comparisons.

2.3 COMPARISON WITH ALTERNATE PROPULSION METHODS

In order to compare pulsed plasma propulsion subsystems with other propulsion methods, weight estimates are made for the other subsystems as outlined below. Other methods qualified for space flight include cold gas and catalytic and heated hydrazine thrusters.[9] Neither cold gas nor catalytic hydrazine systems have significant power requirements, so that their impact on power subsystem design may be neglected.

The propellant weight for these other subsystems may be calculated from:

$$m_p = \frac{M\Delta v}{(32.2)I_{sp}} \tag{10a}$$

where

M = spacecraft mass, lb

Δv = total velocity increment requirements, ft/sec

I_{sp} = specific impulse, sec

Typical cold gas I_{sp} = 72 seconds, and catalytic hydrazine I_{sp} = 215 seconds. Heated hydrazine thrusters, like the ones planned for Intelsat V, have demonstrated about 300 seconds I_{sp}. These heated thrusters require electrical power from the spacecraft.[10] In addition to the propellant allocated for

[9] L. B. Holcomb and D. H. Lee, "Survey of Auxiliary-Propulsion Systems for Communications Satellite," AIAA Paper No. 72-515, April 1972.

[10] R. Grabbi and C. K. Murch, "High Performance Electrothermal Hydrazine (HiPEHT) Development," AIAA Paper No. 76-656, July 1976.

performing the total mission, a propellant reserve of 15% is usually provided. Propellant tankage and pressurant are allotted 13% of the propellant weight for hydrazine liquid storage. Gaseous propellant tankage is heavy, usually equalling the propellant weight.

Thrusters and thruster valves may be allocated 1 pound for each thruster location on the spacecraft. Other dry weight considerations, including the electronics for cycling the thruster valves, are small, and may be neglected for initial subsystem comparisions. For more detailed comparisons, comprehensive parametric information on auxiliary propulsion subsystems may be found in Reference 11 for a variety of propulsive methods.

From the above considerations, it is seen that cold gas systems require the largest propellant allocations, followed by catalytic hydrazine, heated hydrazine, and pulsed plasma propulsion. Thus, significantly more payload mass is potentially available when pulsed plasma propulsion is employed. The example given in Section 3.5 illustrates the weight savings afforded by pulsed plasma propulsion when compared directly with catalytic hydrazine for a typical communications satellite. This example also shows that pulsed plasma propulsion, with the largest power requirements, did not add any weight to the existing power subsystem, but could be accommodated within the existing design.

The cost effectiveness of pulsed plasma propulsion is derived from the added payload capability that can be incorporated in a spacecraft system by virtue of having taken advantage of more efficient propellant utilization. The added weight margin afforded by pulsed plasma propulsion can reduce the number of launch vehicles required, increase system lifetime, or place additional equipment on-orbit. The cost effectiveness for a particular application thus depends on how the margin is allocated.

Cold gas and catalytic hydrazine subsystems have extensive space flight experience. Heated hydrazine is being qualified for use on Intelsat V. Pulsed plasma thruster flight experience is discussed in more detail in Section 5.

[11] L. B. Homcomb, "Satellite Auxiliary-Propulsion Selection Techniques," JPL Technical Report 32-1505, November 1970.

3. INTERFACE REQUIREMENTS

The mechanical, electrical, thermal, and environment interfaces between the pulsed plasma thruster and the spacecraft are described in this section. Mechanical interface data include equipment dimensions, mass properties, and mounting provisions. Electrical data include input voltage and power requirements, isolation and grounding constraints, input command and output telemetry characteristics, and identification of electrical connections on the hardware. Thermal data identify equipment temperature limits and heat dissipation from the equipment when operating. Environmental limits are presented for both nonoperating and operating environments.

3.1 MECHANICAL

3.1.1 Dimensional Interfaces

The millipound pulsed plasma thruster occupies less than 3.5 ft^3 volume, and is approximately 24 inches in diameter and 12 inches high. The power conditioner may be removed from the propellant-discharge assembly and mounted separately. The power conditioner dimensions are 2.5 x 8 x 10 inches.

3.1.2 Mass Properties

The mass breakdown of the pulsed plasma thruster is listed in Table 4. The fully loaded weight is 100 pounds.* The propellant-discharge assembly enclosure is mission-specific. It provides for radio-frequency (RF) shielding of this assembly, passive thermal control coatings, assembly mounting, and radiation hardening, as required by particular applications.

3.1.3 Mounting

The propellant-discharge assembly is mounted via its enclosure, which is mission-specific for the purposes cited above in Section 3.1.2. It is frequently desirable to gimbal mount this assembly to enable thrust vectoring for correcting misalignments, accommodating spacecraft center-of-mass shifts, providing active attitude control torques, or unloading momentum wheels.

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*70,000 lb/sec total impulse capability.

Table 4. Mass Properties

Item	Mass (lb)
Electronic System	8.0
Energy Storage and Nozzle	30.4
Structural	24.1
Propellant (70,000 lb-sec capability)	32
Exhaust Cone	1.8
Miscellaneous	3.7
Total	100.0 pounds

The power conditioner may be mounted inside the propellant-discharge assembly or separately within the spacecraft. It should be mounted from its 8 x 10 inch baseplate to permit a good thermal path for heat dissipation during operation.

3.2 ELECTRICAL

3.2.1 Input Voltage

The thruster requires 28 ± 2 Vdc to the power conditioner.

3.2.2 Input Power

The thruster requires 170 watts at a pulse repetition rate of 1 pulse/5 seconds which corresponds to an equivalent steady-state thrust of 1 millipound. Less frequent pulsing requires proportionately lower average power. Additional power may be required for thermal control or gimbal drives, depending upon particular installations.

3.2.3 Isolation and Grounding

The power lines are grounded in the thruster. The thruster power return (negative terminal of its energy storage capacitor) and the power

conditioner input power return are routed such that they can be grounded at a single point within the spacecraft. Separate power and signal lines are used throughout. The power return and signal returns are separate.

3.2.4 Commands

A 10 Vdc steady state signal is required to command the power conditioner to charge the main energy storage capacitor bank and discharge initiating capacitors. Upon reaching full charge, an internally generated trigger signal initiates the thruster discharge. The process takes 5 seconds for charge and 30 microseconds for discharge. The process repeats as long as the 10 volts is applied.

An optional pulse command may be used for initial start-up. It energizes a single actuation (one shot) latching relay between the spacecraft power bus and the power conditioner.

3.2.5 Telemetry

The following 5-volt analog signals are furnished from the power conditioner:

- Discharge high voltage
- Ignitor circuit input voltage
- Capacitor temperature
- Input voltage to power conditioner.

3.2.6 Electrical Connections

The electrical connections from the spacecraft to the power conditioner area: (1) input bus power-two wires, and (2) command and telemetry signals – seven wires maximum.

The electrical connections from the power conditioner to the propellant-discharge assembly are: (1) primary discharge power and discharge initiating power-four shielded high voltage wires, and (2) telemetry signal - two wires.

3.3 THERMAL

3.3.1 Nonoperating Temperature Limits

Prior to launch, the thruster may be exposed to temperatures from -50° to +50°C.

3.3.2 Operating Temperature Limits

Equipment operating temperature limits are as follows:

Energy storage capacitors	-20° to +50°C
Power conditioner	-50° to +60°C
Other components	-50° to +100°C

3.3.3 Heat Dissipation

Heat dissipation from the equipment, when operating at full thrust, is as follows:

Power conditioner	34 watts
Energy storage capacitors	18 watts
Fuel housing and electrodes	6 watts
Total heat conducted to 40°C mounting plate	58 watts

3.4 ENVIRONMENTAL

3.4.1 Nonoperating Environments

3.4.1.1 Prelaunch

Prior to launch, the equipment may be subjected to the following conditions:

	Minimum	Maximum
Temperature	-20°C	+50°C
Relative Humidity	0	95%
Ambient Pressure	0	1 atmosphere

3.4.1.2 Launch

The equipment may be subjected to Shuttle Transporation System (STS) launch environments.

3.4.1.3 Orbital Operations

In orbit, the equipment is intended to be subjected to a space thermal environment. Nonoperating vacuum and temperature limits are the same as those specified for operating environments (see Sections 3.3.2 and 3.4.2.1).

3.4.1.4 Storage

The thruster should be stored in a container to prevent unnecessary atmospheric contamination and to protect it from unnecessary handling. The thruster may be placed in its container and stored under normal ambient conditions.

A shorting bar should be used during storage to assure that the potentially hazardous energy storage capacitors are fully discharged.

3.4.2 Operating Environments

3.4.2.1 Vacuum

The equipment may be operated at a vacuum of 10^{-4} torr or better.

For ground checkout in air, a simulated thruster load consisting of an ignitron may be attached to the electrodes. The ignitron provides a low impedance load for the capacitor discharge, thus eliminating the need for a Teflon plasma. Ground checkout in air verifies thruster operating voltages and discharge timing.

3.4.2.2 Thermal/Vacuum

The equipment is intended to be subjected to a space thermal environment. Operating temperature limits are given in Section 3.3.2.

3.5 TYPICAL SPACECRAFT CONFIGURATIONS

This section summarizes three Air Force mission applications for pulsed plasma propulsion; DSCS-III, DSP and GSP. The discussions include the propulsion requirements for the pulsed plasma system, the functions performed, and the total weight savings that can be achieved.

3.5.1 DSCS-III Mission

The Defense Satellite Communications System-III (DSCS-III) provides the capabilities needed for effective implementation of worldwide military communications for the next decade. The satellite will be capable of launch from Titan-III or Space Shuttle. For operational use, a constellation of four satellites will be placed in synchronous, equatorial orbit over the Atlantic, East and West Pacific, and Indian Oceans. Table 5 summarizes early DSCS-III design predictions (assumed for considering the application of the millipound pulsed plasma system). The satellite presently uses a monopropellant hydrazine propulsion subsystem.

Figure 11 is an isometric representation of DCCS-III which illustrates the general location of equipment on the spacecraft. Its antenna farm is clustered on the earth pointing (+Z direction) face of the central spacecraft body. The solar arrays are gimballed about the north-south (Y) axis (+Y pointing south) and the hydrazine thrusters are located on the east-west faces of the central body (+X pointing east). Electronic equipment within the spacecraft body is generally mounted from the north and south equipment panels to take advantage of thermal control surfaces (optical solar reflectors) for heat rejection to space.

In order to minimize the time required for orbit maneuvers, it is desirable to have the capability to operate thrusters continuously for several days. The thermal impact for operating a thruster continuously is to increase the thruster assembly radiator area to approximately 120 square inches. In addition, each thruster assembly will require 8 watts standby power to maintain capacitor temperatures above survival limits after the thruster has been off for approximately 8 hours.

Table 5. DSCS-III Design Data Assumed for Study

Design Life	7 - 10 Years
Spacecraft Weight (Dry)	Approximately 1715 lb
Stabilization	0.2° circular error radius, overall accuracy of RF beam axis pointing $\pm 0.1^{\circ}$ orbit positioning accuracy
Attitude Control Subsystem	Accuracy - 0.08° roll 0.08° pitch 0.8° yaw
Electrical Power Subsystem	1100 watts array power (beginning of mission) 837 watts array power (10 years) 28 V $\pm 1\%$
Existing Hydrazine Propulsion Subystem	Thrust levels - 1 lb to 0.3 lb Blow down ratio - 4:1 with full load Specific impulse - 228 sec at initial conditions
Overall reliability	Greater than 0.7 at 7 years

Accordingly, a number of candidate total system configurations can be considered the principal tradeoffs include:

- <u>Boom-Mounted Versus Body-Mounted Thrusters</u>. Boom mounted thrusters (Configuration A in Figure 12) provide for minimum plume interactions with the spacecraft. Body mounted thrusters (Configuration C in Figure 13) provide a lighter weight approach while still maintaining good clearances between the thruster plumes and nearby equipment.

- <u>Spacecraft Rotated 90 Degrees For Orbit Trim and Relocation</u>. Maneuver times are reduced by providing a more favorable thrusting direction for orbit trim and relocation functions by rotating the spacecraft 90 degrees about its Y axis. This reduces the E-W cant angle from 60 to 30 degrees. However, this maneuver does not allow the earth sensor to be used for attitude control.

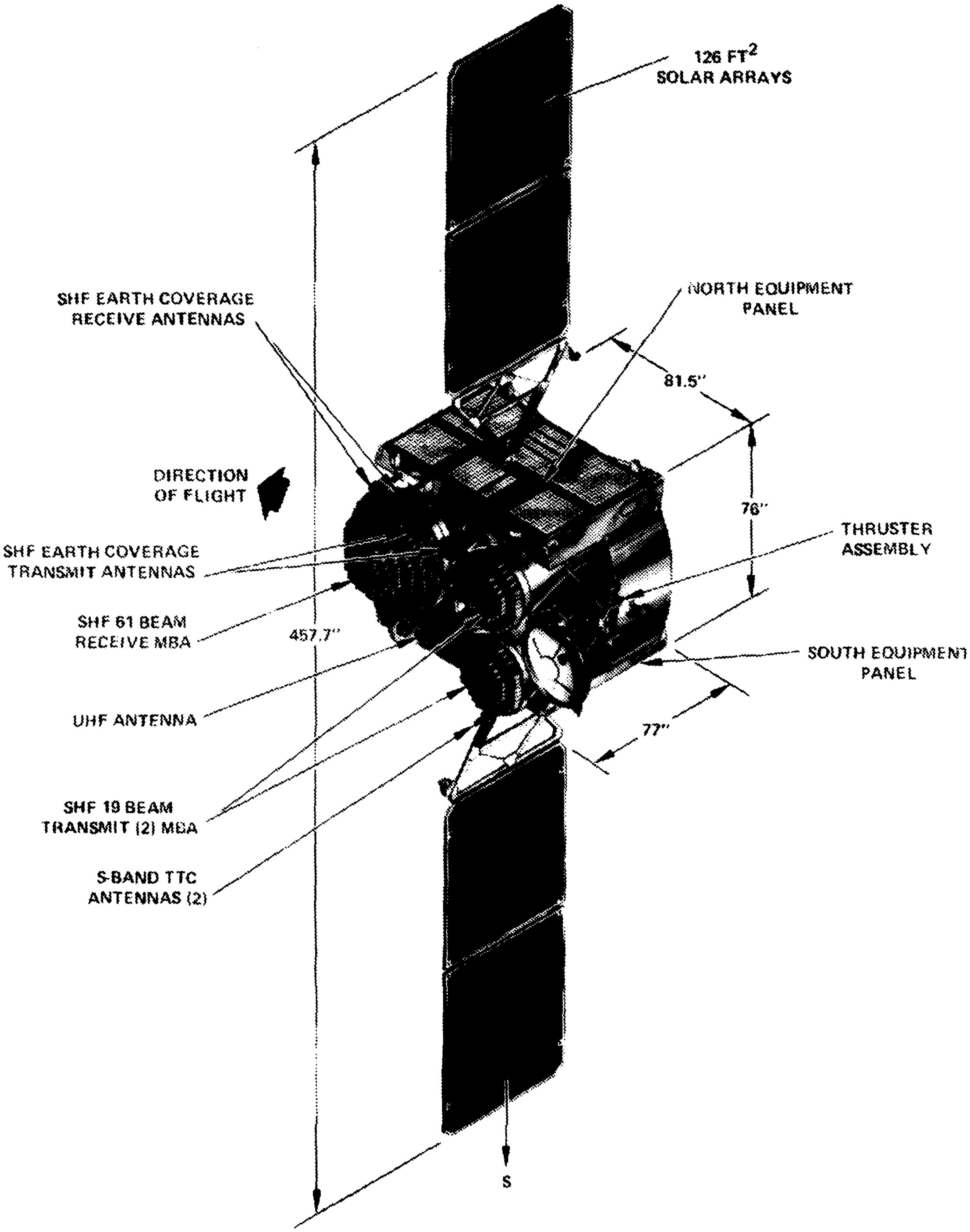


Figure 11. DSCS-III

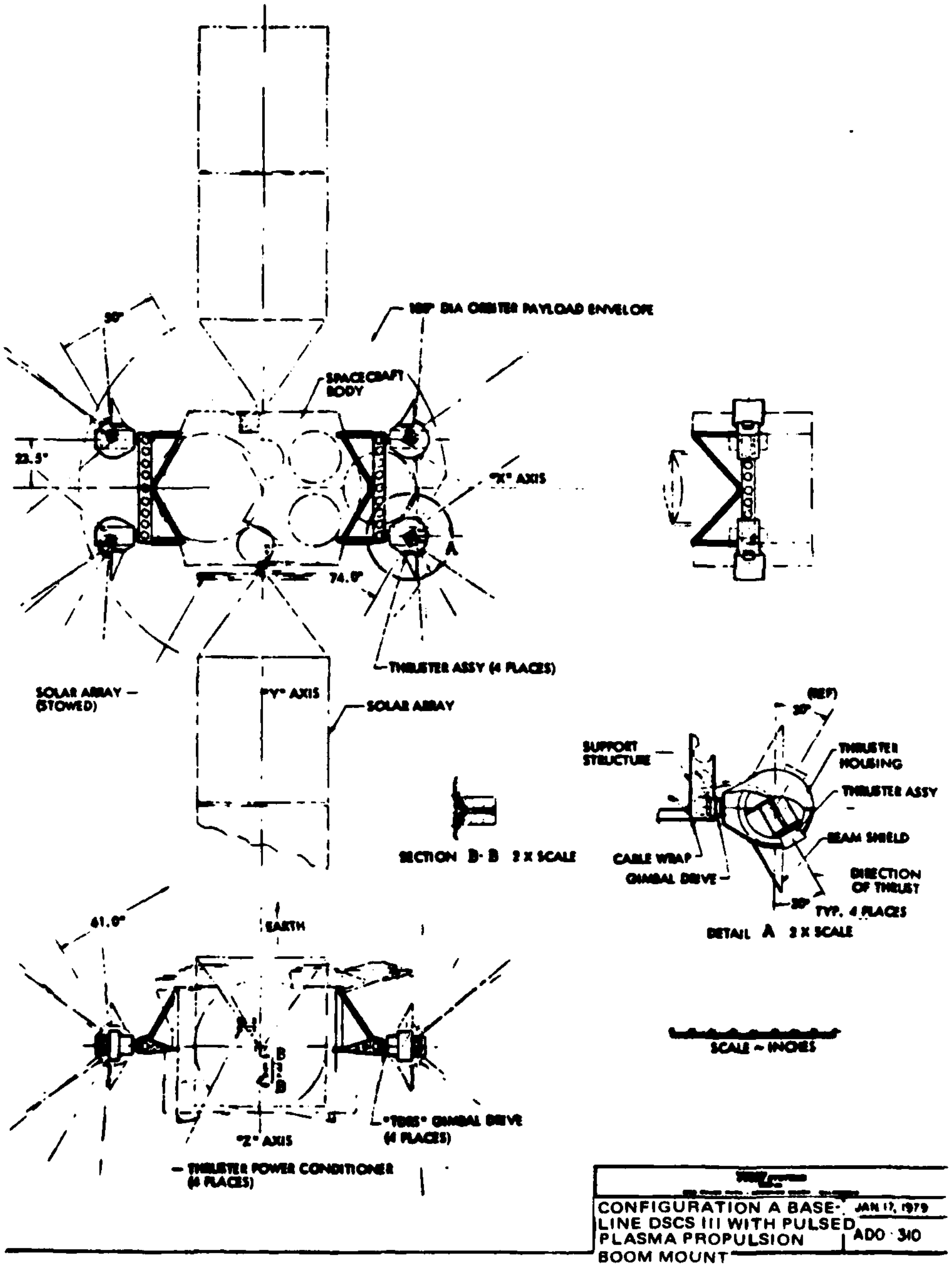


Figure 12. Baseline DSCS-III Configuration with Pulsed Plasma Propulsion

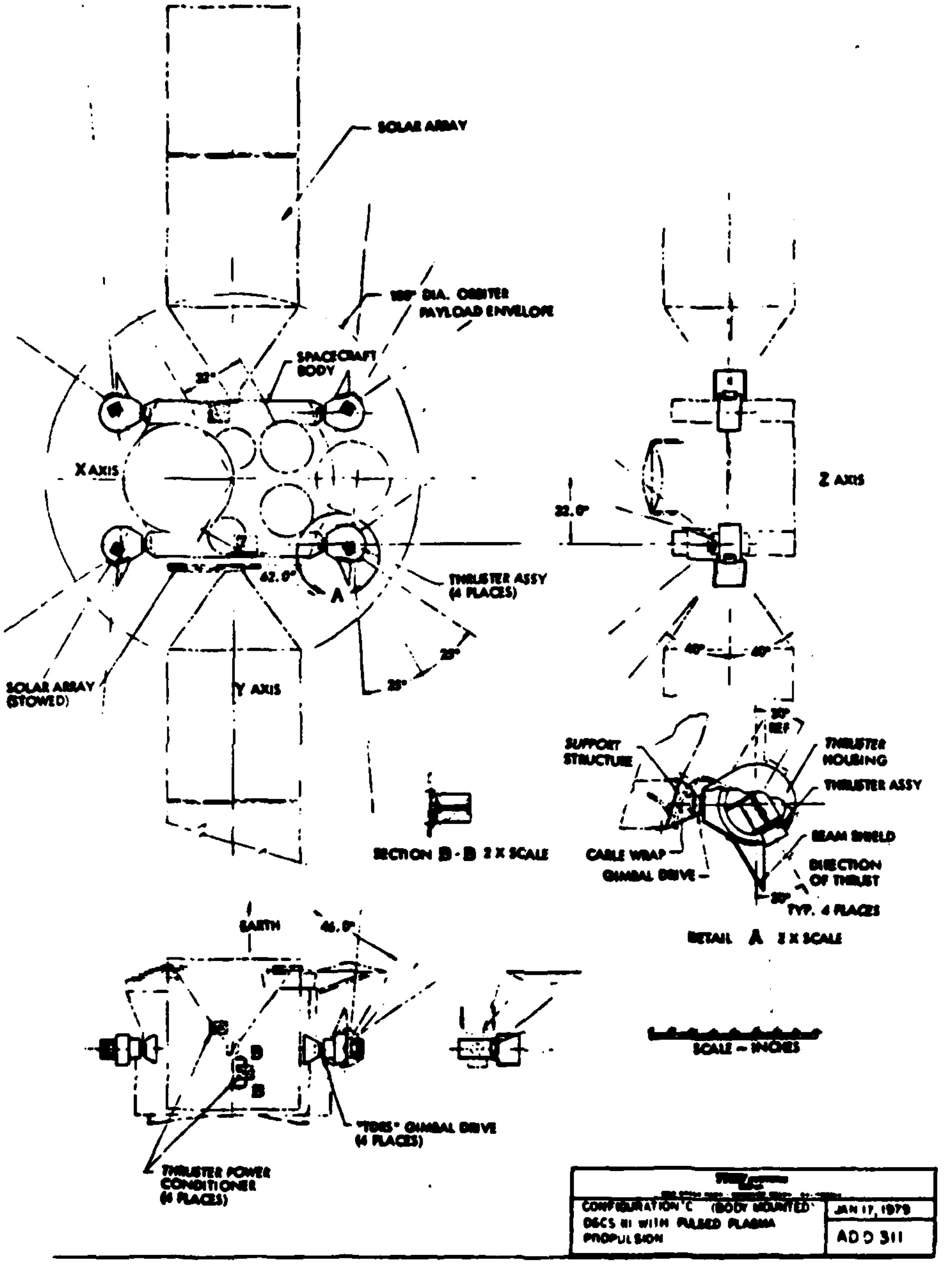


Figure 13. DSCS-III Configuration with Body Mounted Thrusters

- Maximum Number of Thrusters Operating Simultaneously. Maneuver times are reduced by operation at higher thrust levels, i.e., by operating a number of 1 millipound thrusters simultaneously in parallel. The 90 degree spacecraft rotation allows simultaneous operation of all four thrusters for repositioning. Power requirements, however, increase linearly with the number of thrusters thus employed, with attendant impact on the spacecraft electrical power subsystem. This alternative has been designated Configuration B.

- Hydrazine For Orbit Trim and Relocation. Combination hydrazine/pulsed plasma propulsion subsystems yield attractive weight margins, albeit not as large as all plasma propulsion. Hydrazine is used to reduce long maneuver times.

- Active Attitude Control. By using pulsed plasma thrusters for active limit cycle attitude control, the reaction wheels on DSCS-III can be eliminated.

- 245 Days/Year Stationkeeping. By performing stationkeeping functions during favorable seasons when more power is available for propulsion, there is no weight impact on the existing electrical power subystem for DCCS-III.

Table 6 summarizes the results for a 7-year mission for the various tradeoff configurations identified in Table 7. The configurations listed are those of most interest because they illustrate the impact of configurational differences. System weights are for a 50,000 lb-sec capacity Teflon propellant housing.

The propellant weight requirements include a redundancy factor of 2.0 at each thruster location so that the subsystem can tolerate any single point failure. The added power subsystem weight was determined from a detailed analysis of thrust scheduling, available battery power, and excess array capacity as a function of mission life. The results indicate a need for 63.4 watts additional array at EOL (end-of-life) (10 years) for Configurations A and C, and 210 watts BOL (beginning of life) for Configuration B (four thrusters for initial acquisition). Table 6, therefore, shows a net weight of 303 pounds for propulsion on DSCS-III with Configuration A, including propellant redundancy and impacts on attitude control subsystem and power subsystem weight allocations. Also shown for comparison in Table 6 are the times required for several propulsive maneuvers: initial stabilization, orbit trim, and relocation. Other maneuver times are not significantly affected by configurational differences.

Table 6. DSCS-III Tradeoff Comparisons, 7-Year Mission

Configuration	A	A-2	B-2	C-2	C-3	C-4	C-5*	C-6
Maneuver Times, Days (a)								
• Initial Stabilization	1.3	1.3	0.7	1.3	1.3	1.3	1.3	1.3
• Orbit Trim	38	28	17	28	<0.1	28	28	<0.1
• Relocation (b)	13	10	6	10	<0.1	10	10	<0.1
Propulsion Subsystem (Dry) Weight, lb								
• Pulsed Plasma Subsystem	209	208	208	197	195	199	199	197
• Hydrazine Subsystem	---	---	---	---	27	---	---	27
Propellant Weight, lb								
• Teflon (c)	84	82	82	82	76	86	86	80
• Hydrazine	---	---	---	---	25	---	---	25
Attitude Control Subsystem Weight Impact, lb								
• Reaction Wheels	---	---	---	---	---	(22)	(22)	(22)
Power Subsystem Weight Impact, lb								
• Additional Solar Array	10	10	25	10	10	10	---	---
Net Weight for Propulsion, lb	303	300	315	289	333	273	263	307

(a) Assumes average spacecraft weight = 2000 lb

(b) At rate of 0.5 deg/day after full acceleration; maneuver time is for acceleration and deceleration.

(c) Meets 7-year life requirement with any single point failure. Carries > 10 years of propellant with no failures in first 3 years. Propellant housing capacities are 25 pounds each.

*Preferred configuration

Table 7. Candidate DSCS-III Configurations

Identification	Maximum Number of Thrusters Operating Simultaneously	S/C Rotated 90 degrees for Orbit Trim and Relocation	N_2H_4 for Orbit Trim and Relocation	ACS Reaction Wheels Deleted	245 Days/yr Station-keeping
A*	2	-	-	-	-
A-2	2	X	-	-	-
B-2	4	X	-	-	-
C-2	2	X	-	-	-
C-3	2	-	X	-	-
C-4	2	X	-	X	-
C-5	2	X	-	X	X
C-6	2	-	X	X	X

*Baseline

Configuration A-2 is identical to the baseline except that the spacecraft is rotated 90 degrees for orbit trim and relocation maneuvers. (This is discussed more fully in Section 3.2.2.) In this manner, the maneuver time for orbit trim is reduced to 28 days from the baseline time of 38 days.

Configuration B-2 increases the power available to the propulsion subsystem to enable simultaneous operation of four thrusters instead of two. Thus, the maneuver time for orbit trim is reduced to 17 days. The power subsystem impact, however, is 25 pounds of added solar array to support the increased load at beginning of mission.

Configuration C-2 is identical to A-2 except that the thrusters are body mounted instead of boom mounted (baseline mounting). Body mounting of the propulsion subsystem reduces the structural weight requirements by 11 pounds.

Configuration C-3 is a combined hydrazine/pulsed plasma propulsion subsystem. Hydrazine is used for orbit trim and relocation, reducing these maneuvers to less than an hour each. The hydrazine subsystem consists of a single propellant tank, four hydrazine thruster assemblies oriented for in-track thrusting, and associated hardware. The pulsed plasma subsystem is used for the remainder of the propulsive requirements: initial stabilization, stationkeeping, and momentum wheel unloading.

Configurations C-4, C-5, and C-6 eliminate the attitude control subsystem reaction wheels and use pulsed plasma propulsion directly for attitude control. In this manner, 32 pounds of reaction wheels are deleted from the attitude control subsystem. Thus, there is a reduction in the net weight impact for propulsion.

Configuration C-4 is the same as C-2 except for attitude control implementation. C-2 implements reaction wheels. C-4 does not.

Configuration C-5 is identical to C-4 except that stationkeeping is restricted to 245 days/year. In this manner stationkeeping is not performed during solstice seasons, yet inclination control is maintained within ± 0.1 degree. Additional solar array is not required for battery recharging, and there is no weight impact on the existing power subsystem.

Configuration C-6 is also a combination hydrazine/pulsed plasma propulsion subsystem. Maneuver times for orbit trim and relocation are minimized, and stationkeeping is done for 245 days annually to avoid power subsystem impact for battery recharge during the solstice seasons.

As indicated in Table 6, the preferred configuration is C-5. Its maneuver times appear to be of acceptable duration (less than 30 days) and it exhibits the lowest net weight for propulsion. The existing hydrazine propulsion subsystem on DSCS-III requires 358 pounds of propellant to accomplish the baseline 7-year mission. Its dry weight is 77 pounds,(4) thereby yielding a total weight for propulsion of 435 pounds. Table 8 identifies the additional weight margin afforded by the various tradeoff configurations when compared with the existing hydrazine subsystem. It also compares a combination hydrazine/ion propulsion subsystem. This margin may be put to use in the form of additional spacecraft component redundancy, payload capability, etc. The preferred configuration, C-5, shows an additional weight margin of 172 pounds for implementing pulsed plasma propulsion on DSCS-III.

A mercury ion propulsion subsystem, similar to the one described in Reference 7 for an advanced communications satellite, was compared with the pulsed plasma configurations studied for DSCS-III. The ion subsystem exhibits long, and probably unacceptable, maneuver times for orbit trim and relocation. It may require additional thruster redundancy to achieve acceptable reliability. The ion subsystem that is being compared has four thruster complements (thruster, gimbal assembly, propellant reservoir, and power processor per complement). In the event of single thruster failure, north-south stationkeeping is performed at one node only.

The ion subsystem will require 340 watts from a 70 ±20 Vdc bus, 14 watts from a 28 ±1 Vdc bus, and 7 watts at 28 Vdc for each gimbal actuation. This is similar to the 1-millipound pulsed plasma propulsion subsystem electrical interface. Therefore, it was assumed that the impact on the DSCS-III electrical power subsystem would be the same. It was further assumed that there was no thermal impact (active heater power required) for integrating the mercury propulsion subsystem. Thus, there is no additional solar array or battery required for the ion propulsion subsystem provided that north-south stationkeeping is restricted to about 245 days annually.

Table 8. Weight Margin Comparison, 7-Year Mission

Configuration	Net Weight for Propulsion* (lb)	Additional Weight Margin Available (lb)
Hydrazine Propulsion (existing subsystem)	435	0
A	303	132
A-2	300	135
B-2	315	120
C-2	289	146
C-3	333	102
C-4	273**	162
C-5 (preferred configuration)	263**	172
C-6	307**	128
Hydrazine/ion propulsion	281	154

*Includes power subsystem impact

**Includes 22-pound weight savings due to elimination of reaction wheels.

The all mercury ion propulsion subsystem described above weighs about 229 pounds, including 40 pounds of propellant. When compared with the existing hydrazine propulsion subsystem on DSCS-III, it provides an additional 206 pounds of weight margin.

An extended 10-year mission for DSCS-III will require approximately 500 pounds of hydrazine, for a total hydrazine propulsion subsystem weight of 577 pounds. Configuration C-5 therefore affords a weight margin of 314 pounds for the 10-year mission provided that no pulsed plasma subsystem failures occur in the first 3 years. The weight margin for a dual launch equals 2 x 324 = 628 pounds.

In order to provide enough Teflon at each thruster location for 10 years with any single point failure at launch, the total impulse requirement is increased. Table 9 presents the weight margin comparison for a 10-year mission together wih the total impulse capability requirements for each thruster in order to satisfy a requirement of 10-year life with any single point failure. From this table, it may be seen that the preferred configuration, C-5, yields 276 pounds additional weight margin compared with the existing hydrazine propulsion subsystem. It imposes a total impulse requirement of 68,100 lb/sec on each thruster. The added weight margin for a dual launch where configuration C-5 is implemented is then 2 x 275 = 522 pounds.

Pulsed plasma propulsion can be used for initial stabilization, orbit trim, relocation, stationkeeping, momentum wheel unloading, and attitude control functions on DSCS-III. If the time required to perform these functions is acceptable, then it can completely replace the existing hydrazine subsystem.

The pulsed plasma propulsion system (PPPS) schematic for DSCS-III is shown in Figure 14. It is comprised of four thruster assemblies and four power converters. Each thruster assembly is gimbal mounted and contains a thruster, energy storage capacitors, and Teflon propellant. Electrical interconnections between each piece of equipment and the spacecraft is indicated on the schematic diagram.

In order to minimize plume impingement on the solar array during north-south stationkeeping (the function requiring the greatest thrusting time), the thrusters are canted 30 degrees away from the north-south (Y axis) and towards east-west (X axis). The thrusters are oriented with their minimum plume divergence angles in the X-Y plane and fitted with beam shields to further limit their plume divergence.

Each thruster is mounted on a TDRS (Tracking and Data Relay Satellite) gimbal drive to permit rotation about an axis parallel to, and displaced 23.5 inches from the east-west (X) axis. The thruster assemblies are attached to the power converters by flexible cable wraps which permit a 180 degree gimbal rotation.

Table 9. Weight Margin Comparison, 10-Year Mission

Configuration	Net Weight for Propulsion* (lb)	Additional Weight Margin Available (lb)	Total Impulse Required per Teflon Thruster (lb-sec)
Hydrazine Propulsion (existing subsystem)	577	0	--
A	340	237	66,400
A-2	337	240	65,500
B-2	351	226	65,100
C-2	326	251	65,500
C-3	370	207	62,200
C-4	311**	256	68,100
C-5 (preferred configuration)	301**	276	68,100
C-6	345**	232	64,800
Hydrazine/Ion Propulsion	304	273	--

*Includes power subsystem impact

**Includes 22 pounds weight savings due to elimination of reaction wheels

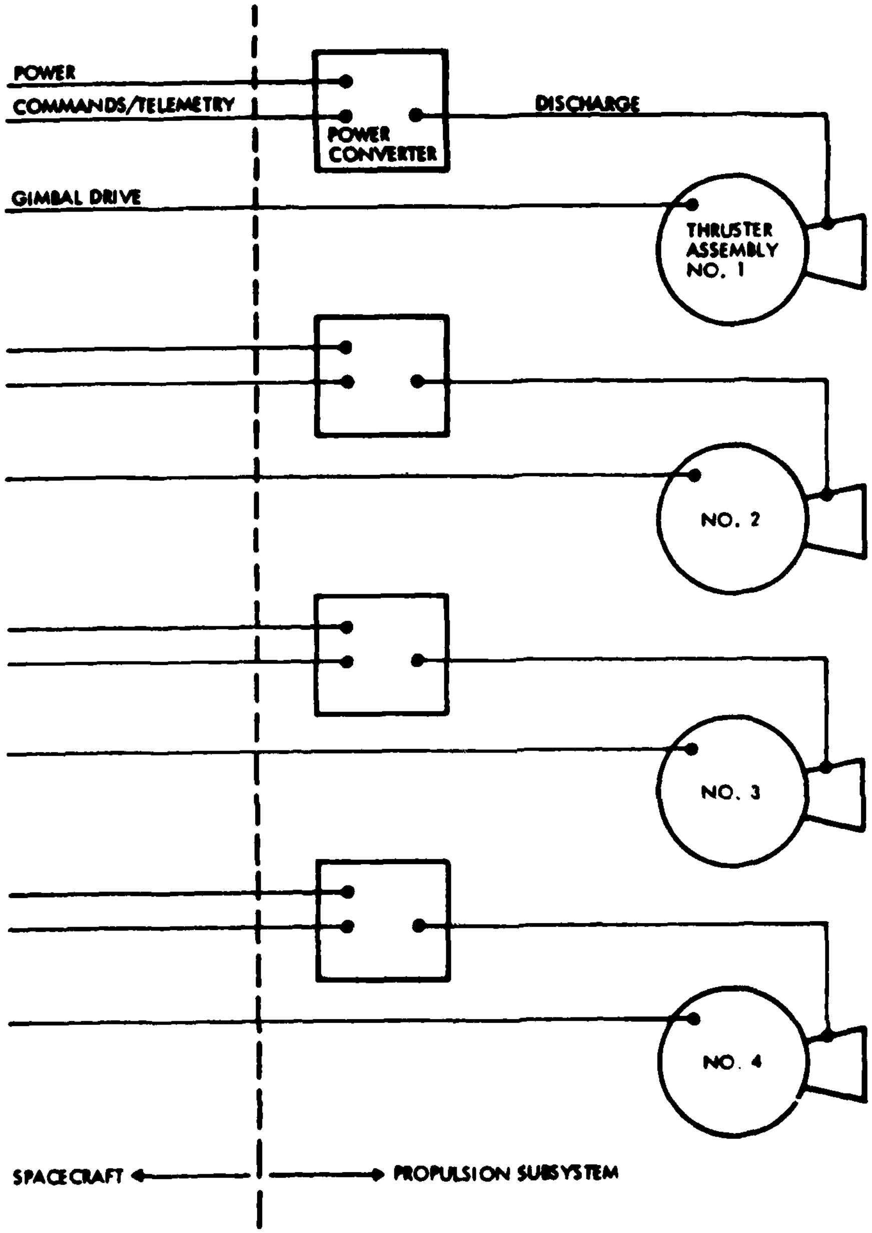


Figure 14. Pulsed Plasma Propulsion Subsystem Schematic for DSCS-III

The thruster support structures extend outward from the central spacecraft body to provide lateral separation of the thruster exhaust from nearby spacecraft surfaces. This lateral separation is constrained by the 180 inch diameter shuttle orbiter payload envelope.

The power converters are mounted separately from the thruster assemblies to simplify propulsion subsystem thermal control. Two converters are mounted on the north equipment panel, and two on the south equipment panel of the central spacecraft body. One converter from each panel is energized when the thrusters operate in pairs. Thus, for example, the converters in the north equipment panel operate the thrusters mounted on the east face, while the converters in the south panel operate those on the west face. The thrusters are located on the east-west faces because the north-south faces are needed for thermal control and the earth pointing face is needed for mounting the antenna frame. The thrusters are gimbal mounted so that they may be reoriented to provide torque control, thrust vector control for specific ΔV maneuvers, or backup capability for other thrusters.

For normal mode operation, two thrusters are nominally pointed north (-y) and two are nominally pointed south (+y). The thrusters are operated in pairs at each nodal crossing for north-south satellite stationkeeping. This permits lateral separation of the thrusters from the solar array axis in order to further minimize plume impingement.

The propulsion power requirement equals (2 mlb) (170 W/mlb) = 340 watts during normal mode thrusting. The propulsion subsystem electrical interface may be tabulated as shown in Table 10, where the power requirements for full thrust operation and gimbal actuation are identified. The total number of commands required is also listed, as is the number of signal channels needed for monitoring thruster status.

Each thruster assembly requires approximately 18 square inches of second-surface mirror radiator area, and the power converters (mounted as shown in Section B-B of Figure 12) will require approximately 36 square inches each on the north and south equipment panels. These results were generated by a conservative transient thermal analysis which calculated radiator areas that can probably be reduced after detailed analysis is performed for flight hardware. The thermal analysis assumed that a pair

Table 10. Propulsion Subsystem Electrical Interface for DSCS-III

POWER	340 Watts at 28 VDC: Full Thrust 10 Watts at 28 VDC for Each Gimbal Motor Actuation		
COMMANDS	FUNCTION	TYPE	NO. REQUIRED
	Thrusting Pulse	Discrete	4
	Gimbal +	Discrete	4
	Gimbal -	Discrete	4
	Total		12
TELEMETRY	SIGNAL	TYPE	NO. REQUIRED
	Discharge Voltage	Analog	4
	Discharge Initiating Voltage	Analog	4
	Capacitor Temperature	Analog	4
	Total		12

of thrusters are operated daily for 2.5 hours at each nodal crossing. Three different power converter installations were considered as follows:

1) Each converter internally mounted within the thruster housing

2) Converter pairs mounted on the east and west faces

3) Converter pairs mounted on the north and south panels

Installation 3 was selected for this study. Installation 1 has the disadvantage of requiring a significant amount of heater power in the survival mode. Installation 2 has the disadvantage of requiring additional radiator area and heater power for the converters mounted on the east and west panels. Installation 3 was selected as optimum for DSCS-III because continuously operating electronic components are located on the north-south panels, hence the required radiator area could be achieved by simply opening up the existing radiator. A single node analysis was performed on a thruster for a worst case transient heatup of 2.5 hours with direct solar input into the radiators and a transient cool down to the minimum temperature limit of -60°F.

3.5.2 DSP Mission

The Defense Support Program (DSP) satellite is shown schematically in Figure 15. Table 11 lists the major design characteristics assumed for the study which is based on a future version of the satellite currently planned. The auxiliary propulsion requirements are dictated by the fact that the satellite flies in an earth synchronous equatorial orbit and spins at 6 rpms around its earth pointing (+Z) axis. In order to precess this spin axis around in orbit without excessive propellant demands, a retrograde momentum wheel is added to give the satellite zero momentum. Electrical power is provided at all times of the day by a distributed solar cell system consisting of a conical segment, cylindrical segment, and four two-way solar paddles. Batteries provide power during an eclipse. Because of the special characteristics of the DSP satellite, three areas of concern have been identified which are of special interest for pulsed plasma propulsion: (1) EMC/EMI, (2) particulate contamination, and (3) optical radiation.

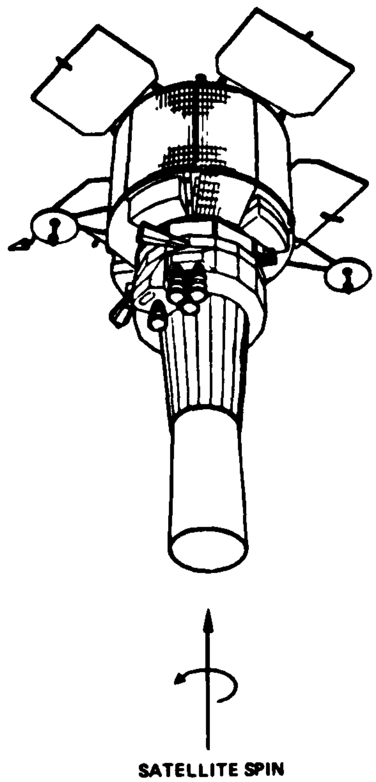


Figure 15
Schematic Diagram of DSP

Table 11. DSP Design Data Assumed for Study

DESIGN LIFE	7 Years
SPACECRAFT WEIGHT (DRY)	3200 lb (352 lb hydrazine capacity)
STABILIZATION	Tight Pointing Accuracy for Spin Axis 2° Orbit Positioning Accuracy BOL 1° Orbit Positioning Accuracy EOL
ATTITUDE CONTROL SUBSYSTEM	Zero Total Angular Momenum Satellite; Main Body Spin Rate 6 ± 0.225 rpm Around Yaw Axis
ELECTRICAL POWER SUBSYSTEM	750 watts BOL 23 watts Margin 31.8 ± 0.2 volts Noneclipse 24.5 to 32.0 volts Eclipse Season
EXISTING HYDRAZINE PROPULSION SUBSYSTEM	Thrust Levels - 3.5 lbf High Level Attitude Control and ΔV 0.039 lbf Low Level Attitude Control

Figure 16 illustrates the basic layout of the hybrid propulsion system. The hydrazine subsystem is comprised of a central tank and four relatively high level thrusters to provide lateral ΔV and torque capability. The pulsed plasma subsystem consists of three thruster systems placed in line on the outer shell of the spacecraft. Figures 17 and 18 show the detailed installation concept; Figure 19 shows the installation concept for the hydrazine subsystem.

The outermost (primary) pulsed plasma thrusters (1 and 3 in Figure 16) provide the required propulsive capability. The middle thruster (2 in Figure 16) is included for redundancy. The primary thrusters are displaced 27.8 inches to either side of the plane through the spacecraft CG. They can be fired in tandem to provide a horizontal ΔV in any desired direction. Roll/pitch torque control can be provided by either commanding a phased delay between the two thrusters or firing only one of the two primary thrusters. The primary thrusters are gimballed about axes parallel to the spin axis to provide yaw (spin) control. The gimbal axis for the redundant thruster is parallel to the Y axis in order to allow a choice of roll/pitch polarity depending on which thruster is being replaced. When all three thrusters are operational the spare thruster can be used to enhance system flexibility.

The propulsion functions performed include initial stabilization, orbit trim and stationing, relocation, east-west and north-south stationkeeping, and attitude control.

3.5.2.1 Initial Stabilization

After tip-off, the satellite goes through a rather conventional acquisition in that it uses wide field-of-view sun sensors to partially stabilize the satellite. It rotates about the sun vector to find the earth and is then three axis stabilized. The satellite is spun up and orbit trim is performed. Though undesirable, the low torque of the pulsed plasma thrusters is acceptable for sun acquisition. Any two thrusters (with their gimballing capability) provide three axes of torque control. The earth acquisition requires a higher level thruster since the earth search mode is made at a reasonably high search rate so that the earth can positively be acquired during a limited search window in orbit. The

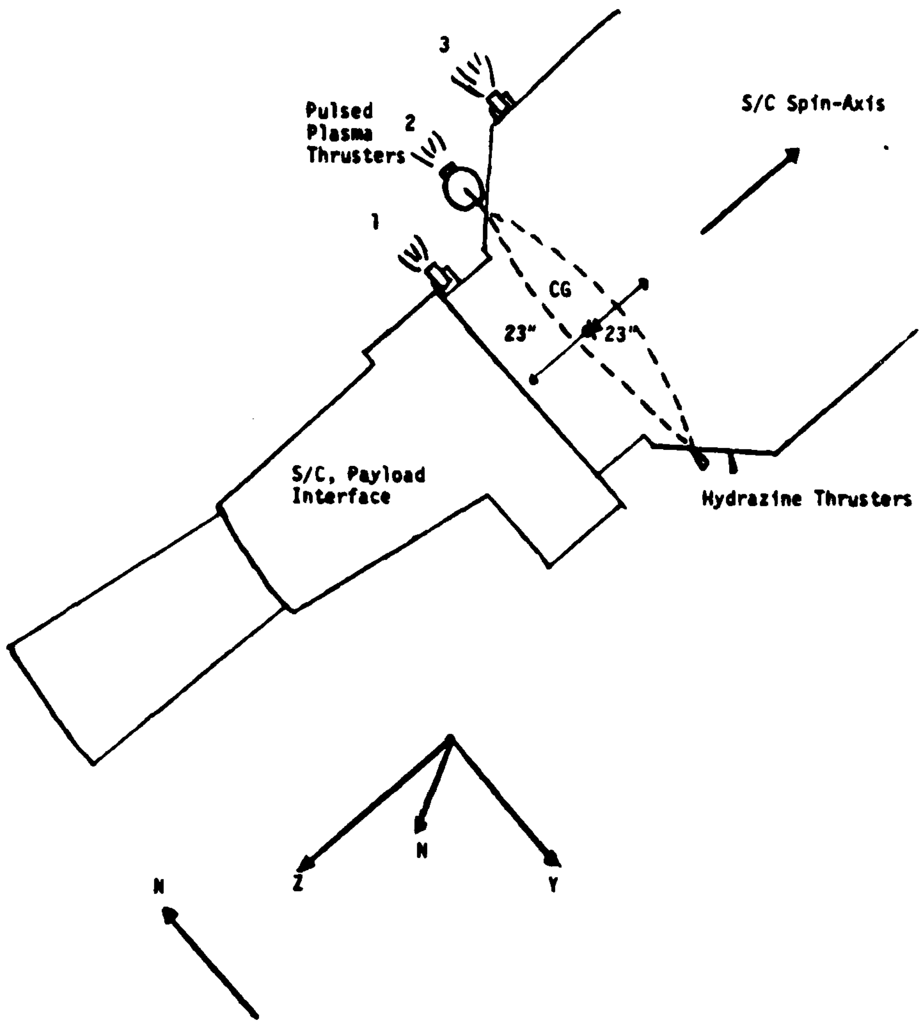


Figure 16. DSP Thruster Configuration

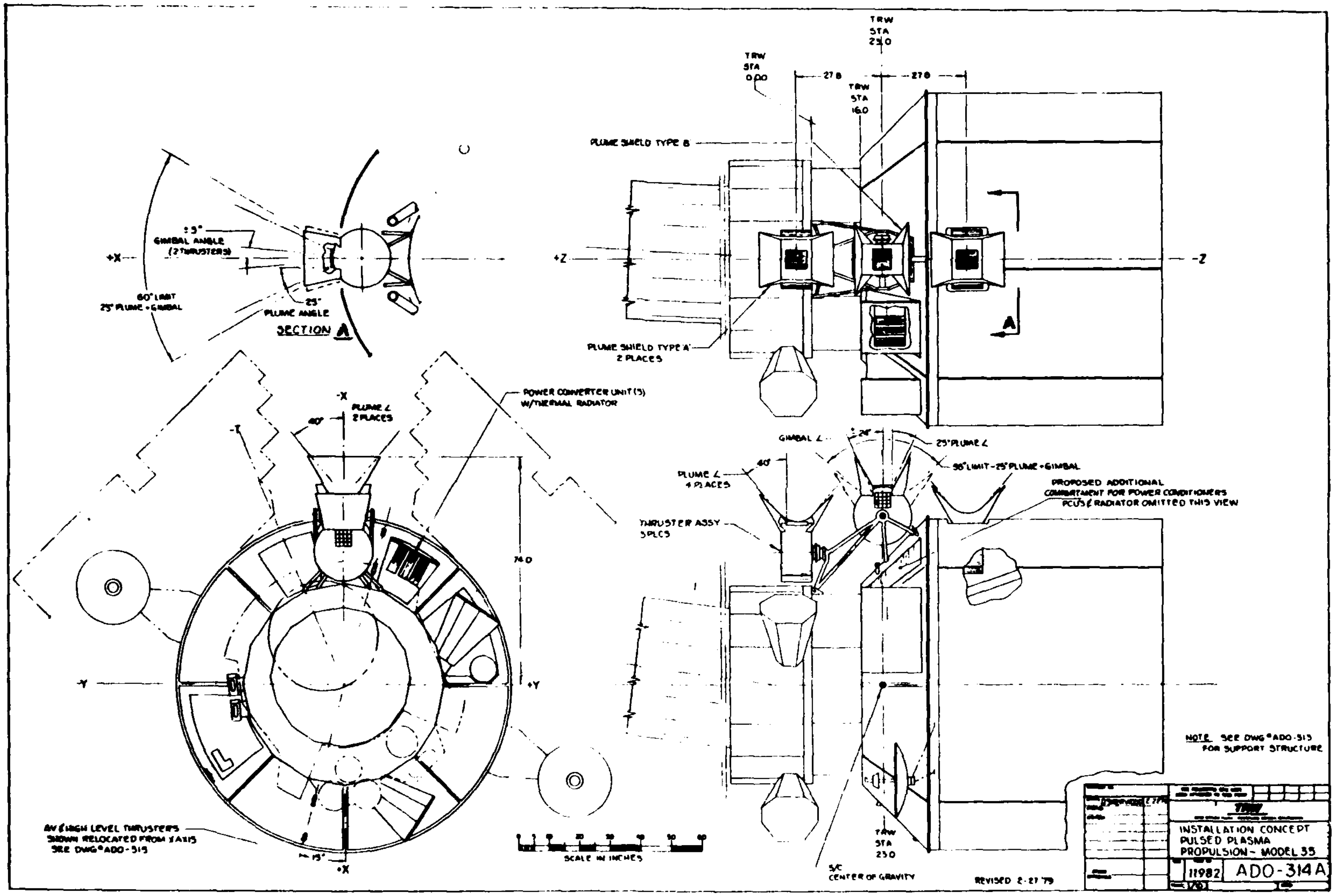


Figure 17. DSP Installation Concept

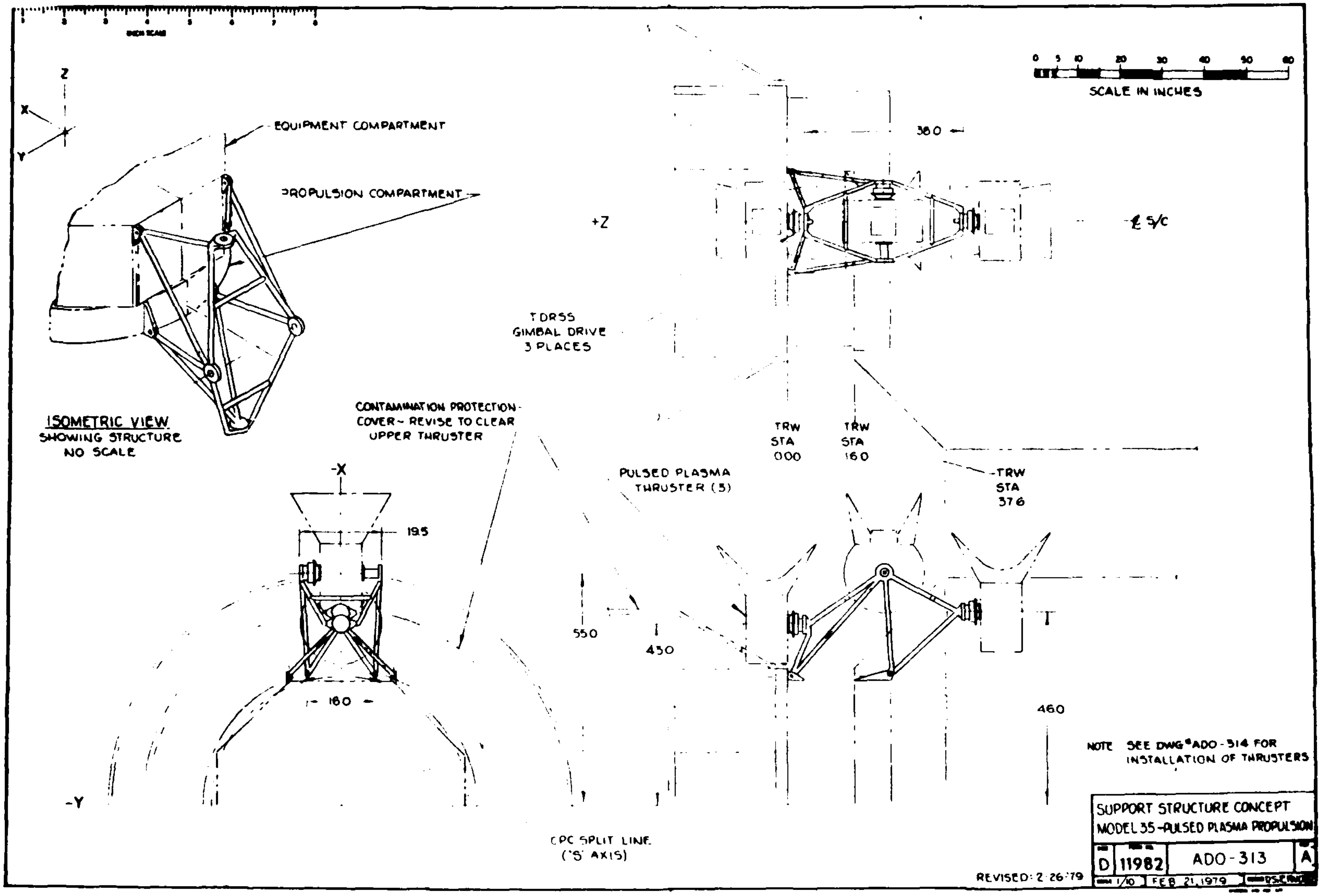


Figure 18. DSP Support Structure Concept

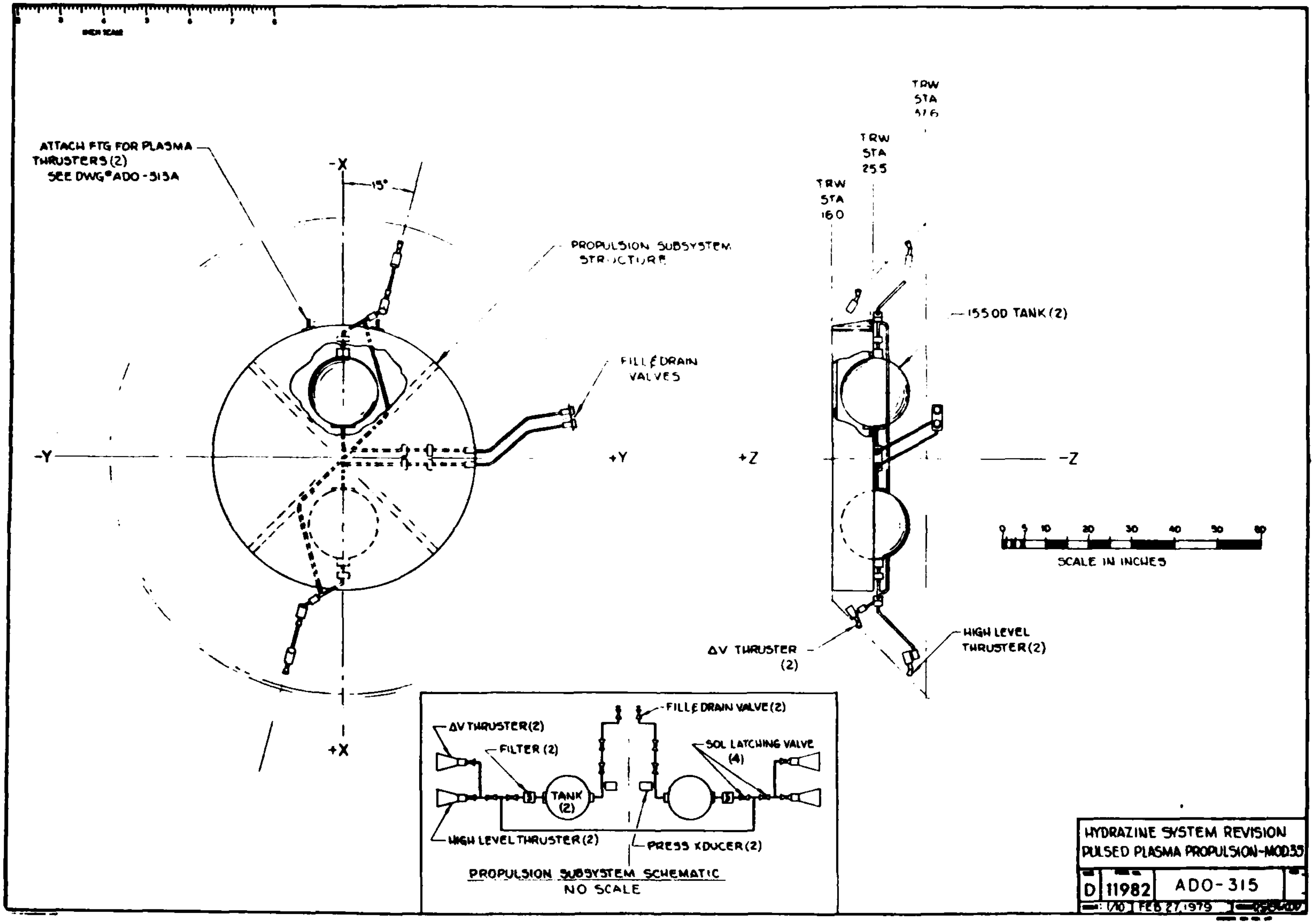


Figure 19. DSP Hydarazine System Revision — Pulsed Plasma Propulsion

high level thrusters are required to acquire the earth within the limited field of view of the earth sensor (~18 degree arc). Other search acquisition schemes could be employed that would permit the low level pulsed plasma thrusters were hydrazine not required for change of station. Since hydrazine would be available, it is a small cost and weight penalty to use and retain a proven acquisition technique.

3.5.2.2 Orbit Trim and Stationing

After attitude stabilization has been achieved, a longitudinal correction must be made to move the satellite from the injection station to its desired station and correct out any residual drifts. The total velocity correction is estimated to be 28 ft/sec. This can be performed either by the pulsed plasma or hydrazine system. Each thruster produces a pulse each rotation when the thruster is parallel to the desired thrust direction. It is likely that a hydrazine system would require 14 days to complete, firing all three thrusters once per rotation, and would require 300 watts of average power. The pulsed plasma thrusters would be used to maintain attitude control during hydrazine firing.

3.5.2.3 Relocation

There is the potential need to change the satellite station once in orbit. If required, the change of station could be as much as 180 degrees and must be completed within 14 days. Impulsively, this would require 120 ft/sec ΔV capability and the pulsed plasma system could only provide 28 ft/sec within this time frame. This is the predominant requirement for a hydrazine capability.

3.5.2.4 Combined E-W, N-S Stationkeeping and Attitude Control

These are the primary propulsive functions for which the pulsed plasma thrusters are well suited. In the interests of minimizing propellant and average power requirements, a strategy has been developed which simultaneously performs each of these thrust functions.

The thrusters have been arranged so that any combination of a north or south ΔV, and east or west longitude ΔV, stationkeeping (but not repositioning) and any desired combination of three axes of torque control can

be provided with a timed pair of thruster pulses. Although use of thruster 3 (see Figure 16) increases the flexibility, the following discussion will assume use of thrusters 1 and 2. Similar arguments can be made with use of 1 and 3 or 2 and 3.

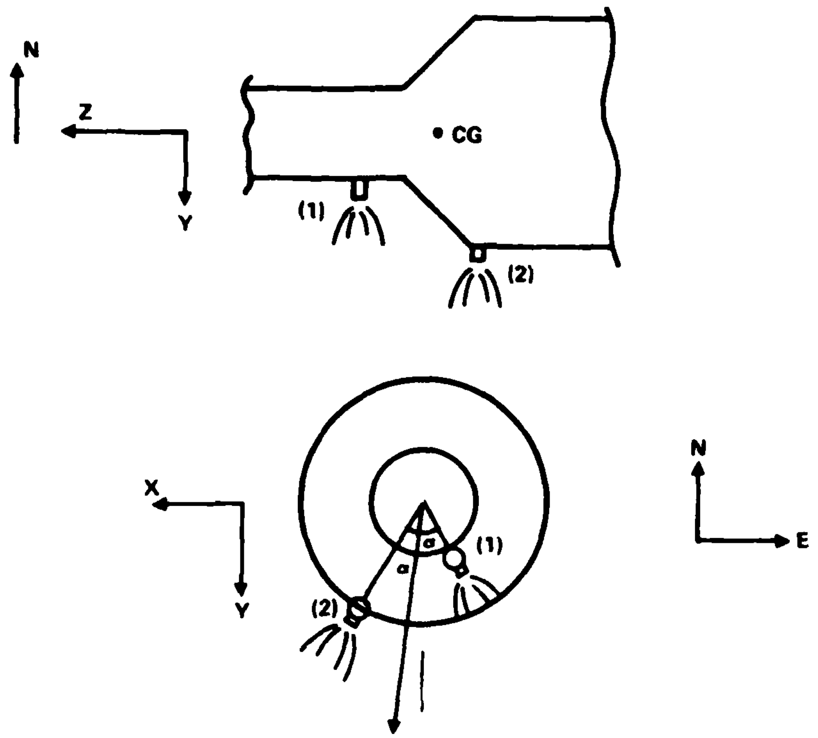
The major propellant consumption is due to north-south stationkeeping. To achieve a north increment, firing thrusters 1 and/or 2 while their thrust vectors are in the southern half of the satellite rotation (Figure 20). Peak efficiency is achieved at the south pointing direction and the efficiency degrades as a cosine function of the angle away from the location. By symmetry, the opposite angles are used for south stationkeeping.

To combine one east component with the north stationkeeping, the pulse pair is delayed the appropriate amount to allow the satellite to produce an east component of velocity. Conversely a west component is generated by speeding up the pair of pulses.

A torque about the spin axis (Z or yaw) is created by gimballing the thrusters so they both, or at least one, does not have its thrust vector passing through the spacecraft center of mass. A torque about the other two axes is produced by either firing just one thruster when its thrust vector is normal to the desired torque axis; or the two thrusters are staggered in time, producing a torque about the axis bisecting the two firings. These are idealized situations, assuming each thruster pulse is identical in terms of Δangular momentum. Actual misalignments create second order disturbances which must be controlled out at some future time.

The control algorithm to create these composite maneuvers consists of simple and logical decisions. When a pulse is commanded for either stationkeeping or attitude control, the timing is selected to contribute towards the other factor. For example, if a Y (pitch) torque is required to correct for solar disturbance torques, a rapid scan is made to see whether north or south and east or west stationkeeping is called for. The satellite is allowed to rotate to the point where both a pitch torque and appropriate stationkeeping are provided. If south stationkeeping is called for, thruster 1 is fired when it points north.

The basic stationkeeping accuracy requirement is $\leq$ 1.0 degree. However, for an all hydrazine system, in the interest of reducing propellant weight,



NET STATIONKEEPING VECTOR DIRECTION AND TORQUE AXIS

Figure 20. Example of Combined N, E Stationkeeping Plus Yaw Attitude Control Pulse

the DSP north-south stationkeeping strategy is to initially incline the orbit with a 2-degree bias relative to the plane of the equator. The orbital plane is then allowed to freely rotate for approximately 2 years without north-south stationkeeping, until the orbit plane is nearly geosynchronous. At this time stationkeeping is employed until all the fuel is expended (approximately 1 year before end of mission). The orbit plane is then allowed to incline 1 degree during the last year of the mission. The EOL criteria are tighter than BOL since the sensor will then be degraded. The degradation is partially compensated by tighter stationkeeping. This particular strategy would not be optimally employed for pulsed plasma thrusters since thrusting is eliminated at the beginning of the mission, a time when excess electrical power is readily available. A more optimum strategy is to use the early surplus power to maintain an off-geosynchronous bias and allow gradual drifting as less power becomes available. This maintains tight pointing at EOL and minimizes added power demands.

Thus, the basic strategy is to thrust as close to the orbital nodes as possible and exploit as much early life excess power as possible to minimize both propellant and electrical power consumption. These two goals are somewhat contradictory since concentrating thrusting at the nodes requires a high average electrical power and spreading the thrust cycling about the entire day causes inefficient propellant utilization. In addition to this basic quandary, there are other constraints to consider. The basic redundancy approach has been to use two thrusters at one time, holding the third in reserve until a failure occurs. This way, only two are available for firing at any time. Since the spacecraft rotates at 1 revolution/10 seconds, two thruster pulses per 10 seconds is the highest average stationkeeping thrust available. When the pre-empting of these thrusters on the average of once every 62.3 seconds for ACS is factored in, the average stationkeeping thrust level becomes 0.84 mlb when 170 watts of power are available for stationkeeping and ACS. Another problem arises in attempting to make full use of the thrusters at the very beginning of life since at that time sufficient power exists to over-correct and actually precess the orbit plane away from geosynchronous orbit. Finally, the average electrical power available fluctuates seasonally as well as with orbit life. Figure 17 shows these fluctuations. Peak excess power occurs in winter when the sun is nearer the spacecraft. Least excess power exists at the equinoxes since power is required to charge the batteries in preparation for solar eclipses.

In developing a thruster strategy, the following factors were considered:

1) Launch was assumed at winter (worst case)

2) Power excess dictated average thrust level available per season

3) A nominal thrust schedule of two 8-hour segments each day was assumed (where possible)

4) Added lifetime average power requirements were calculated to achieve ~880 ft/sec ΔV over a 7-year period

With this strategy only 45 watts of additional array power capability is required for pulsed plasma application. It should be noted that this is

possible because thrusting is greatest when excess power is greatest. At end-of-life eclipse periods, in order to put minimum strain on the electrical system, pulsed plasma thrusting levels are low. Note also that 15 watts of the 45-watt increase is required just to bring current DSP 5-year lifetime up to the new 7-year mission duration. The data are summarized in Table 12. With 45 watts added power, the desired 880 ft/sec was not achieved. However, the delivered 856.4 ft/sec permitted end-of-life inclination of -0.13 degree, which is certainly satisfactory. With a thruster failed at launch, the inclination at EOL is -0.6 degree.

The total impulse required from the pulsed plasma thrusters for north-south stationkeeping, taking orbital inefficiencies into account for thrusting ~8 hours at each node, equals 87,100 lb/sec. To this is added 22,380 lb/sec required for ACS. Thus the total impulse required for ACS and stationkeeping equals 109,480 lb/sec. This can be provided with 50,000 lb/sec thrusters if the thruster provided for redundancy at BOL is employed near EOL to furnish 9480 lb/sec for achieving inclination of -0.13 degree at EOL in absence of any primary thruster failures.

Table 13 presents the DSP propulsion subsystem comparison resulting from the configuration studies. The combination pulsed plasma/hydrazine subsystem provides for beginning-of-life (BOL) satellite inclination error of 1 degree and end-of-life (EOL) inclination of 0.13 degrees. When compared with the basic hydrazine subsystem, which provides for 2 degrees BOL and 1 degree EOL, the combination pulsed plasma/hydrazine subsystem exhibits a weight advantage of 131 pounds. When compared with a hydrazine subsystem having equal capability, it exhibits a weight advantage of 340 pounds. Increased accuracy for inclination control at EOL is desirable to accommodate sensor degradation. As indicated in Table 12, 1 degree EOL is acceptable for the basic mission.

3.5.3 GPS Mission

The NAVSTAR Global Positioning System (GPS) is a space based radio frequency navigation system which provides signals for accurate determination of position, velocity, and system time by properly equipped users.[12] The

[12] Navigation, Vol. 25, No. 2, Summer 1978.

Table 12. North-South Stationkeeping Impulse Summary (with 45 W addition to electric power subsystem)

YEAR	SEASON	POWER AVAILABLE FOR ACS & N-S ΔV (w)	AVERAGE THRUST AVAILABLE FOR N-S ΔV (mlb)	N-S ΔV FROM ACS FIRING (ft/sec)	N-S ΔV AT NODAL FIRINGS (ft/sec)	TOTAL N-S ΔV	CUMULATIVE ΔV (ft/sec)
1	WINTER	275	.84	7.2	49.7	56.9	56.9
	SPRING	130	.64	2.3	13.7	16.0	72.9
	SUMMER	170	.84	7.2	52.3	59.5	132.4
	FALL	100	.47	2.3	10.1	12.4	144.8
2	WINTER	200	.84	7.2	55.9	63.1	207.9
	SPRING	80	.35	2.3	7.5	9.8	217.7
	SUMMER	135	.67	7.2	44.0	51.2	268.9
	FALL	75	.32	2.3	6.9	9.2	278.1
3	WINTER	185	.84	7.2	57.8	65.0	343.1
	SPRING	70	.29	2.3	6.2	8.5	351.6
	SUMMER	120	.59	7.2	38.8	46.0	397.6
	FALL	70	.29	2.3	6.2	8.5	406.1
4	WINTER	175	84	7.2	57.8	65.0	471.1
	SPRING	65	.26	2.3	5.6	7.9	479.0
	SUMMER	120	.59	7.2	38.8	46.0	525.0
	FALL	60	.23	2.3	4.9	7.2	532.2
5	WINTER	165	.84	7.2	55.9	63.1	595.3
	SPRING	55	.20	2.3	4.3	6.6	601.9
	SUMMER	110	.53	7.2	34.8	42.0	643.9
	FALL	50	.17	2.3	3.7	6.0	649.9
6	WINTER	155	.79	7.2	51.9	59.1	709.0
	SPRING	45	.14	2.3	3.0	5.3	714.3
	SUMMER	100	.47	7.2	30.4	38.1	752.4
	FALL	40	.11	2.3	2.4	4.7	757.1
7	WINTER	145	.73	7.2	49.0	55.2	812.3
	SPRING	35	.09	2.3	2.0	4.3	816.6
	SUMMER	95	.44	7.2	28.9	36.1	852.7
	FALL	30	.06	2.3	1.4	3.7	856.4

Table 13. DSP Propulsion Subsystem Comparison, 7-Year Mission

Configuration	Hydrazine (Basic MIssion)	Combination Pulsed Plasma-Hydrazine	Hydrazine (Expanded mission)
Orbit Inclination Error, Degrees			
BOL	2	1	1
EOL	1	0.1*	0.1
Dry Weight, Pounds			
Hydrazine Subsystem	99	30	127
Pulsed Plasma Subsystem	--	155**	---
Propellant Weight, Pounds			
Hydrazine	353	66	534
Teflon	---	68	---
Total Subsystem Weight, Pounds	452	321	661
Δ Weight, Pounds	131		340

* With no single point failure in first 5 months. With a failure at launch, EOL inclination error = 0.6 degree.

**Using 50,000 lb/sec capacity systems.

navigation signals will be provided by a constellation of 18 three-axis stabilized satellites uniformly distributed around three 12-hour orbits, each inclined 55 degrees relative to the equator. This configuration will allow any user, anywhere on earth and at any time, to be within line-of-sight of at least four satellites as requied for determining three components of position and system time. Implementation of the system is currently scheduled in three phases as indicated in Figure 21.[13] The Phase I space segment consists of eight navigation development satellites (NDS). The goal is to maintain a minimum of five on orbit at all times. The Phase II space segment calls for an additional three satellites to be launched on an as-needed replenishment basis to maintain six satellites on station. The 18 satellite Phase III space segment is scheduled for initial operational cabability in the mid-1980s.

The pulsed plasma thruster can be used as part of an autonomous control loop to minimize in-track position error for Phase III satellites. At the present time, in order to maintain knowledge of the satellite location to the desired accuracy, the GPS ground segment performs satellite tracking and control functions which include daily orbit calculation. In addition, the GPS is limited in its ability to predict satellite location within a 24-hour period. Random solar radiation pressures are the primary contributors to this limitation which amounts to in-track error of $>$ 2 meters.

A pulsed plasma propulsion system can be used in conjunction with a single axis Disturbance Compensation System (DISCOS) sensor to automatically compensate the in-track component of solar pressure, thus eliminating the GPS dependence on daily ground station updating.

[13] B. D. Elrod and A. Weinberg, "NAVSTAR GPS," Navigation, Vol. 25 No. 3, Fall 1978, pp. 323-344.

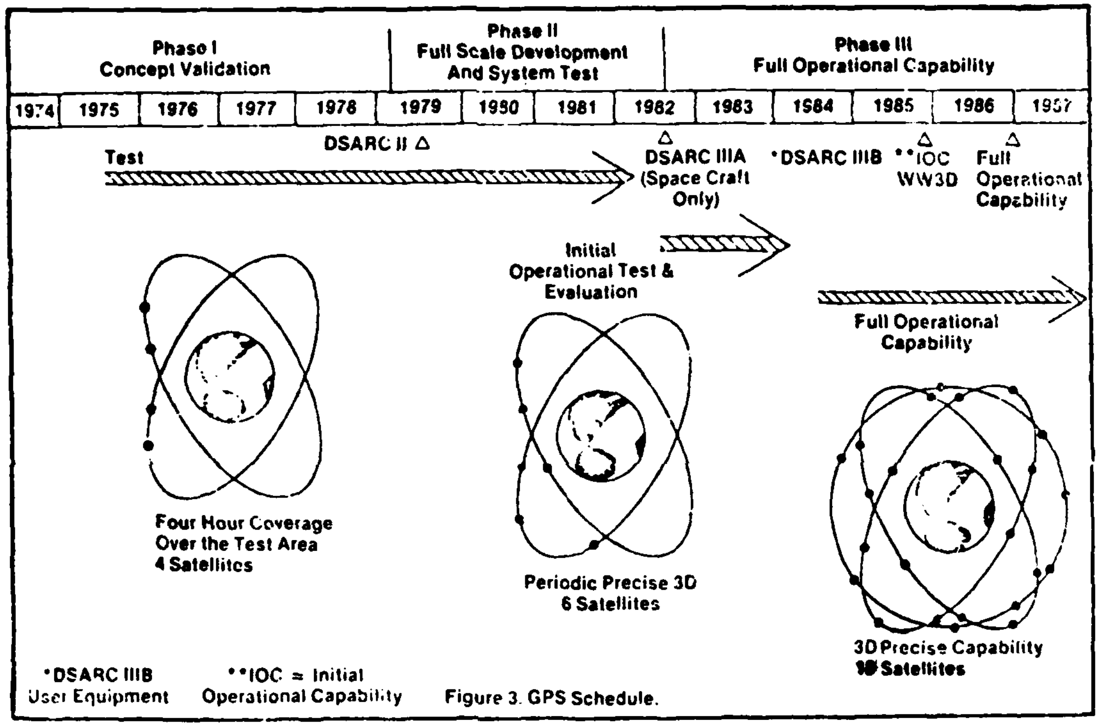


Figure 21. Overview of NAVSTAR GPS (Reference 13)

The major mission parameters are summarized in Table 14. Figure 22 shows the presently planned Phase III design.[(14)] As will be seen later, several major changes are required to accommodate the DISCOS navigational system.

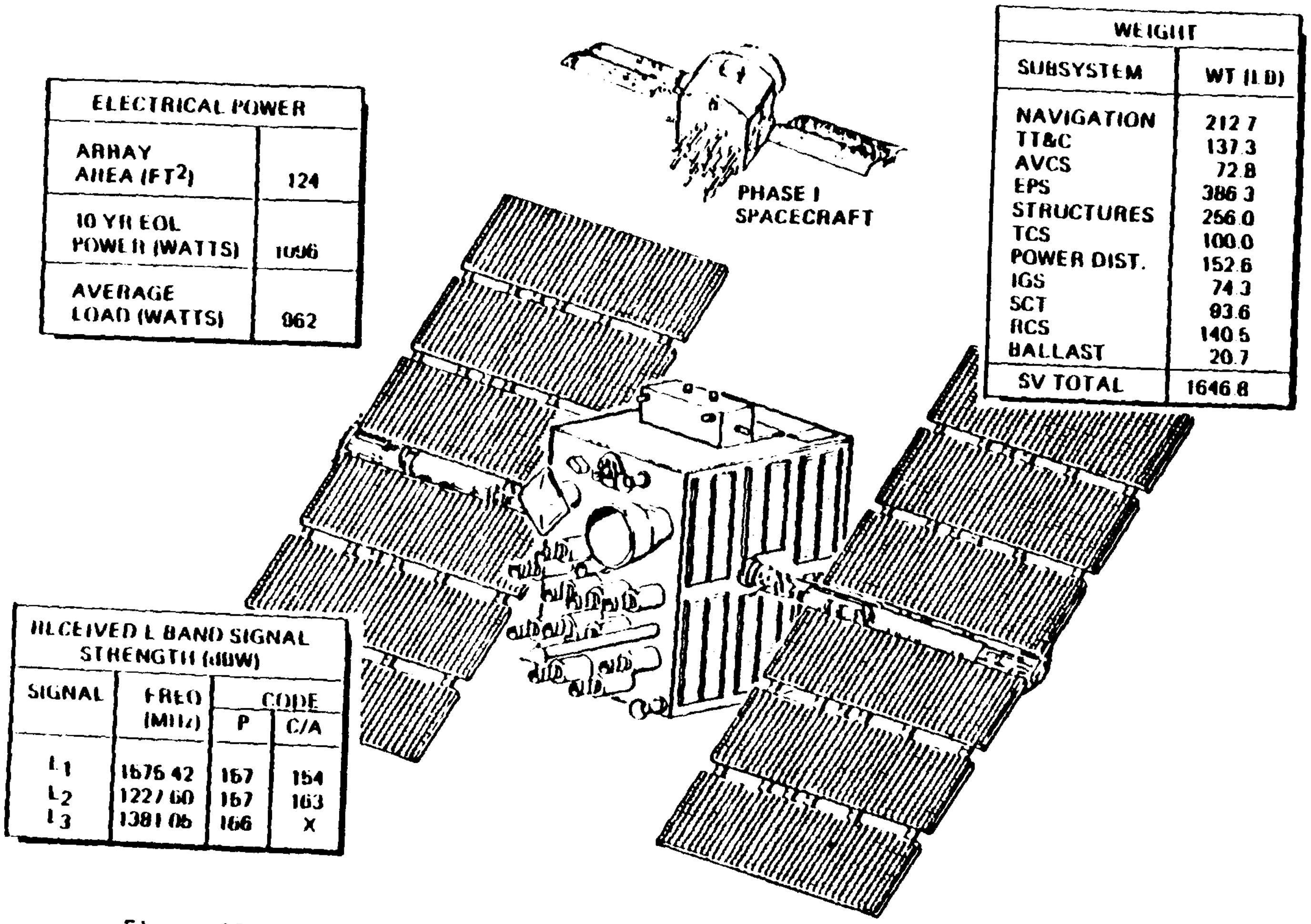
Table 14. Global Positioning System Mission

Payoff with Pulsed Plasma Propulsion	Improved position accuracy
Orbit	Half-synchronous, circular
Altitude	10,900 nautical miles
Inclination	55 degrees
Period	12 hours
Mission Life	10 years
Spacecraft Power Load (end of life)	1000 watts
Satellite Weight	∿1650 pounds
In-Track Position Accuracy	<1 meter

The GPS satellites use monopropellant hydrazine for performing the following maneuvers

- Spacecraft spin (0-25 rpm) and despin
- Velocity changes in the spin mode
- Reorientation of the spin axis (precession)
- Three-axis control and stabilization
- Velocity changes under three-axis control

[14] "GPS Phase III Space Vehicle Systems Engineering Study," Final Report, Rockwell International, 27 April 1979.



ELECTRICAL POWER	
ARRAY AREA (FT^2)	124
10 YR EOL POWER (WATTS)	1096
AVERAGE LOAD (WATTS)	962

RECEIVED L BAND SIGNAL STRENGTH (dBW)			
SIGNAL	FREQ (MHz)	CODE P	CODE C/A
L_1	1575.42	157	154
L_2	1227.60	157	163
L_3	1381.05	166	X

WEIGHT	
SUBSYSTEM	WT (LB)
NAVIGATION	212.7
TT&C	137.3
AVCS	72.8
EPS	386.3
STRUCTURES	256.0
TCS	100.0
POWER DIST.	152.6
IGS	74.3
SCT	83.6
RCS	140.5
BALLAST	20.7
SV TOTAL	1646.8

Figure 22. Baseline GPS Phase III Space Vehicle Design (Ref. 14)

The subsystem contains 53 pounds of propellant operating in a blowdown mode. The 16.5-inch diameter tanks provide potential propellant loading up to 121 pounds while maintaining thruster inlet pressures of 90 psia at end of mission. The basic maneuvers will continue to be provided by hydrazine. The pulsed plasma system will be added solely for the DISCOS application.

The basic idea of the DISCOS is that a tiny satellite is completely enclosed in a cavity of a larger satellite. The inner satellite, or proof mass, is shielded from all external surface forces, drag, and radiation pressure. As part of the design, it is necessary to:

- Eliminate all possible force interactions, including mass attraction between the two bodies
- Sense the relative displacement between the two bodies
- Use this displacement to modify the motion of the (outer) satellite. As a result, the satellite is constrained to the orbit of the proof mass, which is free of all external forces

Pulsed plasma thrusters have been used on navigation satellites in conjunction with DISCOS sensors to provide for autonomous drag compensation. This concept was used on Johns Hopkins Applied Physics Laboratory TIP II and TIP III satellites.[15] A single-axis DISCOS (Figure 23), as described in Reference[16], was integrated with fore and aft firing thrusters for realizing drag compensation. An identical arrangement is being implemented by RCA for the NOVA spacecraft series.

The 55-degree orbit inclination imposes severe configuration/attitude control requirements for maintaining proper orientation of the solar arrays and thermal control surfaces. This is because the 55-degree orbit plane may be inclined by as much as 78.5 degrees with respect to the ecliptic.

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[15] W. J. Guman and S. J. Kowal, "Pulsed Plasma Propulsion System for TIP-II Satellite," JANNAF Propulsion Meeting, 1975.

[16] F. F. Mobley, G. H. Fountain, A. C. Sadilek, P. W. Worden, Jr. and R. Van Patten, "Electromagnetic Suspension for the TIP-II Satellite," IEEE Transactions on Magnetics, Vol. MAG-II, No. 6, November 1975, pp. 1712-1716.

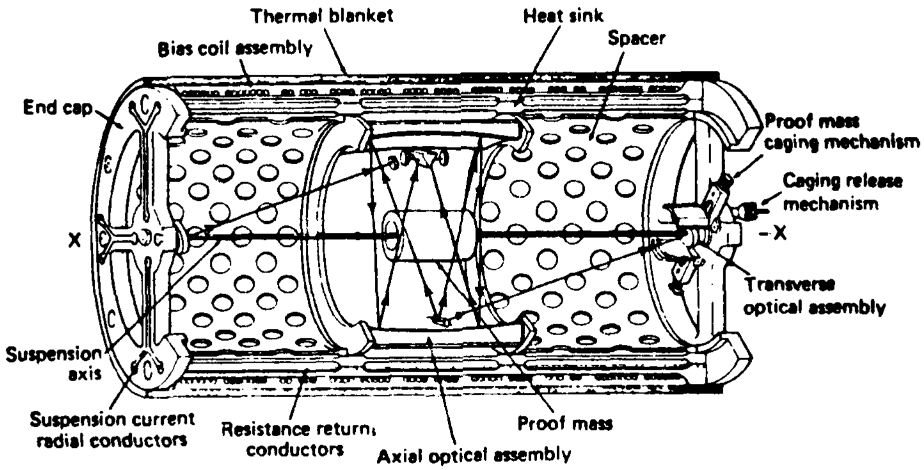

Figure 23. Single-Axis DISCOS Concept

By way of comparison, a synchronous equatorial orbit is inclined by only 23.5 degrees with respect to the ecliptic plane. The significantly higher declinations of the GPS orbits require essentially a two degree of freedom solar array control and additional control maneuvers for avoiding solar incidence on thermal control surfaces. The present NAVSTAR concept provides the additional degree of freedom and attitude control by seasonal yaw maneuvers. A similar strategy for a DISCOS system would be undersirable because of the requirement to maintain the DISCOS axis and thrust direction parallel to the velocity vector.

Figure 24 shows the layout for the GPS/DISCOS version. To provide complete redundancy four thrusters are located in pairs on the east and west satellite faces, in line with the satellite center of mass. In order to accommodate DISCOS, the following design modifications would have to be implemented:

- Two-degree-of-freedom solar array. The present GPS concept utilizes a single axis solar array drive combined with seasonal yaw ›neuvering to maintain proper sun orientation for the planned 55 degree inclination orbits. The DISCOS modification uses a second of freedom for the solar array boom in order to avoid the yaw requirement. This is necessary to maintain the DISCOS sensing and thrusting axis (i.e., the satellite X axis) collinear with the velocity vector at all times. The required array articulations are described in Figure 24.

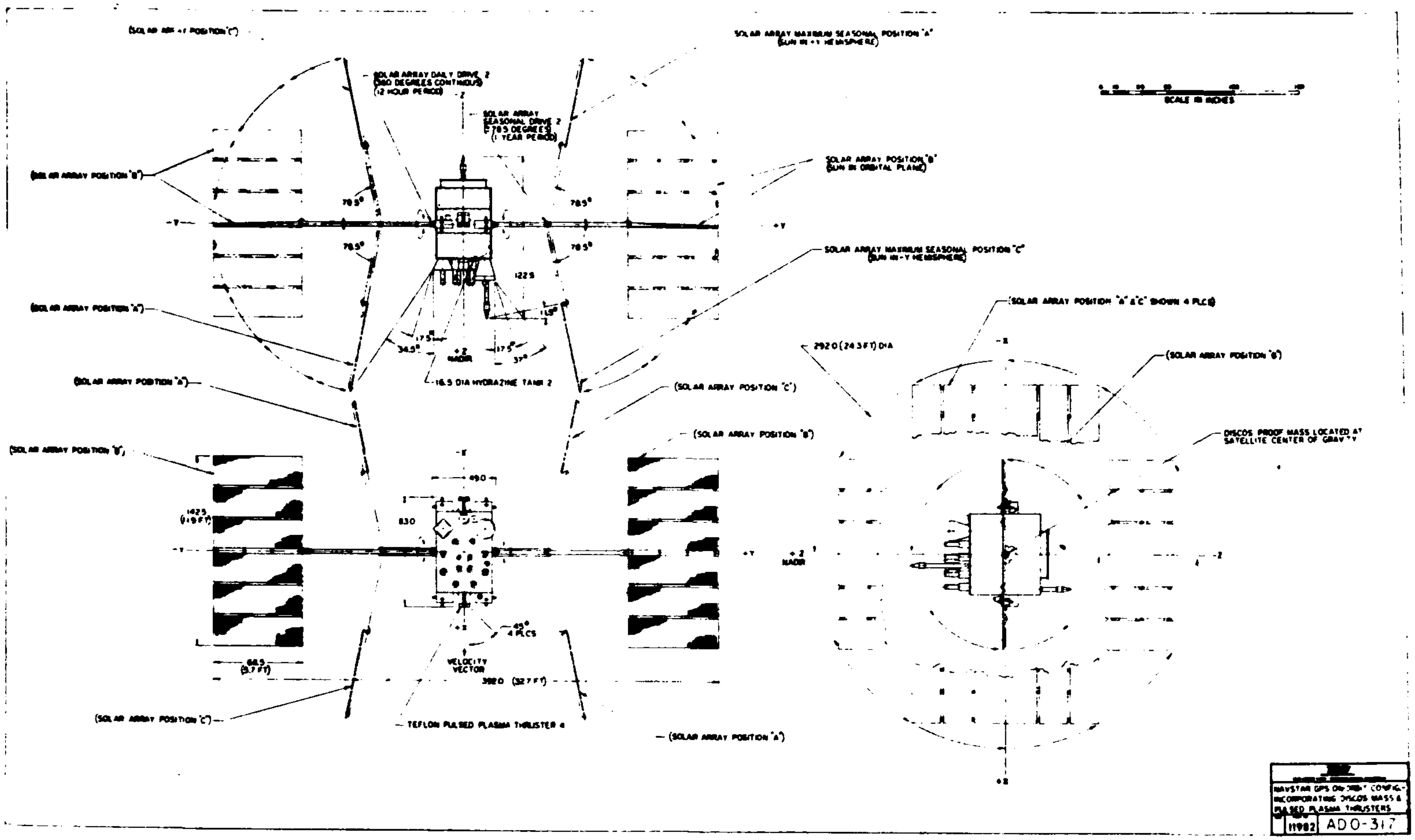

Figure 24. NAVSTAR GPS on-Orbit Configuration Incorporating DISCOS Mass and Pulsed Plasma Thrusters

The array booms are sized, articulated, and driven so as to avoid shading each other or the spacecraft and to maintain constant centers of gravity and pressure. To accomplish this, the main array drive rotates the array with a 12-hour period about the y axis. The seasonal drive provides yearly back and forth articulation of the array boom hinge to keep the array perpendicular to the sun line. The maximum articulation ($\pm \Theta_m$) required is equal to the declination of the orbital plane. A semiannual 180-degree yaw maneuver can be employed to limit the articulation to one quadrant (i.e., $0 \le \Theta \le \Theta_m$). The yaw maneuver will also allow a thermal control surface that does not receive full sun incidence.

The overall configuration keeps the array outside a 45-degree cone half-angle with respect to the thruster exhaust. Figure 24 presents further details of the solar array configuration.

- Existing hydrazine thrusters moved to new locations. This was necessary to allow the DISCOS thrusters to be mounted in-line with satellite CG.

- Redistribution and symmetrization of mass distribution. A volume of approximately 4 inches in diameter by 8 inches in length (along X axis) must be made available at the satellite CG. The satellite mass must be distributed to reduce mass attraction forces on DISCOS to $<10^{-8}$ g in the Y-Z plane. In order that the gravitational forces remain sufficiently low throughout the range of DISCOS travel, it is also necessary that the gradient of mass attraction forces be less than 10^{-11} g/mm along the X axis.

- Redesigned satellite thermal control. In addition to the usual thermal control requirements for battery and electronics, GPS Phase III will have a thermal control requirement of ± 0.1°C for its Cs and Rb clocks.

  The present Phase III design uses yaw maneuvers to maintain the solar vector in the X-Z plane. This allows the Y faces to be used for thermal control since they never receive direct sun. Since the DISCOS GPS will not yaw, all external spacecraft surfaces will receive a nearly one sun solar flux during some period of an orbit. Thus more radiator area (and potentially more heaters) will be required. It is possible to provide one surface that never receives direct sun by performing a 180-degree maneuver twice a year, at local noon, when the sun is in the orbit plane. This maneuver will also reduce the total annual solar array reticulation by 50%.

- Possible redesign of antenna farm. The biaxial solar array drive allows the array to penetrate the forward hemisphere relative to the plane of the antenna farm (+Z face). Although the arrays remain well out of the direct line of transmission, more detailed study is needed to determine the impact on antenna radiation patterns. If interference exists, further redesign will be needed to narrow the unidirectional beam patterns and it may be necessary to relocate the S band omnidirectional antenna.

Table 15 summarizes the propulsion requirements for the GPS/DISCOS application. Note that the maximum allowable impulse bit for efficient thrusting is 1.8 mlb/sec, which is less than the 5 mlb/sec delivered by the millipound pulsed plasma system. For this reason a smaller system, an upgraded version of the earlier LES-9 system(17) is the recommended choice. The characteristics of the upgraded LES-9 thruster are shown in Table 16. To guarantee no single point failure will require a total of four thruster assemblies, resulting in a total system weight of approximately 68 pounds. Table 17 compares the total system characteristics with those of more conventional gaseous nitrogen or hydrazine gas generator systems for performing the same function.

Table 15. Propulsion Requirements for GPS/DISCOS

Total Impulse	3200 lb-sec
Impulse per Thruster	1600 lb-sec
Number of Thrusters	4 (2 primary - 2 redundant)
Impulse Bit	$\leq 1.8 \times 10^{-3}$ lb-sec per pulse $\geq 2 \times 10^{-4}$ lb-sec per pulse
Duty Cycle	$\leq 1\%$
Number of Pulses	889,000 pulses

[17] B. A. Free, et al, "Electric Propulsion for Communications Satellites," AIAA 78-537, Communications Satellites Conference, San Diego, Calif. 1978

Table 16. Upgraded LES-9 Pulsed Plasma Performance for GPS/DISCOS

Discharge Voltage	3000 volts
Discharge Energy	80 joules
Impulse Bit	340 μlb-sec
Specific Impulse	1450 sec
Total Impulse	3400 lb-sec
Number of Pulses	10^7 pulses

Table 17. Propulsion System Comparison for GPS/DISCOS Application Candidate System

Parameter	Teflon Pulsed Plasma	Gaseous Nitrogen	Hydrazine Gas Generator
Total Impulse (lb-sec)	3200	3200	3200
Impulse Bit (mlb-sec)	0.34	1.8	1.8
Number of Pulses (Total/All Thrusters)	9.4×10^6	1.8×10^6	1.8×10^6
Specific Impulse (sec)	1450	65	105
Propellant Weight (lb)	4.4[a]	50	31
System Weight (lb)	68	132	47[b], 54[c]

a - Includes 100% propellant redundancy

b - Based on using excess tankage capacity already available on GPS

c - Assumes 7 lb additional dedicated tankage and valving

4. INTERACTION CONSIDERATIONS

There are certain design constraints imposed on the location and operation of pulsed plasma thrusters to avoid undesirable interaction with the spacecraft. The information presented below identifies these constraints so that a spacecraft designer can take them into account when integrating pulsed plasma thrusters into his designs.

4.1 EFFLUX COMPATIBILITY

The exhaust from the solid Teflon millipound pulsed plasma thruster consists of an accelerated plasma slug traveling at an average velocity of roughly 20,000 m/sec and neutral effluent traveling at considerably lower velocity. The plasma slug contains high velocity carbon and flourine ions. Surfaces placed in the path of the accelerated plasma will be subjected to sputtering erosion. Quite generally, Langmuir probe data, collimated quartz crystal microbalance (QCM) data, and data taken of the energy distribution of the plume wake indicate that the energetic plume is located within a cone angle of ±40 degrees about the geometric centerline of the nozzle.[18] The low velocity region extends out to wider divergence angles and consists of neutral carbon and fluorine atoms plus carbon-fluorine radicals. It also contains dilute amounts of eroded electrode and spark igniter materials.

Diagnostic probe measurements of the 1 mlb thruster plume at a distance of 40 cm from the exit plane of the exhaust cone showed that the energetic plume is almost rectangular in cross section.[19] The plume is broader in the direction of the electrodes. This is not surprising because the interelectrode spacing is about 10 times the spacing between fuel bars at the propellant exit plane, and the aspect ratio of the cone exit plane is about 1.5. The rectangular shape of the plume extremities appears to be more influenced by the cone since their aspect ratio is closer to 1.5 than

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18 W. J. Guman and M. Begun, "Pulsed Plasma Plume Studies," AFRPL-TR-77-2, March 1977.

19 D. J. Palumbo and M. Begun, "Experimental and Theoretical Analysis of Pulsed Plasma Exhaust Plumes," AFOSR-TR-78-1242, 1 April 1977-31 May 1978.

10. Deposition pattern measurements at a distance of 76 cm from the exit plane showed the plume to be essentially axisymmetric[20] at that location.

For typical interactive effects analyses, the permissible buildup of material on spacecraft surfaces is only on the order of 10^{16} molecules/cm^2 (approximately 10 monolayers). For an auxiliary propulsion mission this is usually about one part in 10^9 of the material released by the thruster. It is convenient to introduce a normalized efflux parameter, ε, which is defined as the rate of accumulation of material per unit area at the spacecraft surface divided by the plume flux released by the thruster. Thus, the usual statement of plume measurement requirements is that they proceed to the $\varepsilon = 10^{-9}$ cm^{-2} level. Most plume data to date define the thrusting characteristics of the exhaust and extend to approximately the $\varepsilon = 10^{-5}$ cm^{-2} level. There is, therefore, a need to extend the sensitivity of plume efflux measurements downward by about another four orders of magnitude to at least the 90-degree divergence angle for an unshielded thruster, and preferable, into the backward hemisphere as well.

Beam shields have been used effectively with ion engines to intercept wide angle effluent and avoid impingement on sensitive surfaces.[21] Beam shields should also be effective with pulsed plasma thrusters. If the thruster is shielded, material transport measurements must also extend into the penumbra (partially shielded) and umbra (totally shielded from line-of-sight with the propellant exit plane) regions created by the beam shield.

Measurements of material accumulation from the pulsed plasma thruster were taken using either deposition plates or QCMs.[18, 22] These measurements suffered from the effects of test facility presence and determined that facility effects are of dominant concern in the high divergence angle regimes which must ultimately be examined. The backscatter

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20 L. K. Rudolph, L. C. Pless, and K. G. Harstad, "Pulsed Plasma Thruster Backflow Characteristics," AIAA 79-1293, June 1979.

21 S. Zafran, ed., "Ion Engine Auxiliary Propulsion Applications and Integration Study," NASA-CR-135312, July 7, 1977.

22 L.C. Pless, L.K. Rudolph, and D.J. Fitzgerald, "Plume Characterization of a One-Millipound Solid Teflon Pulsed Plasma Thruster," AFRPL-TR-78-63, October 1978.

flux from plume impingement on the test facility walls, even in the Molecular Sink Facility (MOLSINK) at Jet Propulsion Laboratory (JPL), was too large to allow a simple direct measurement of plume back flow from the thruster.

In order to fully characterize the thruster's exhaust plume, the AFRPL is sponsoring a program at the Arnold Engineering Development Center (AEDC) in which plume species and their velocities will be measured. Quantitative mass flux measurements will be made in the back flow, side flow, and core regions with emphasis on the back flow region (<90 degrees from the thruster centerline).

Recommended diagnostic instruments for pulsed plasma thruster plume measurements include directly heated Langimuir probes and Faraday cups. These instruments are specifically intended to focus attention upon the charged particle efflux during thrusting. Detection of material released in an ionized state is straightforward and can be performed with much greater sensitivity (approximately 5 orders of magnitude) than for comparable quantities of un-ionized material. The sensitivity level of Faraday cups with sufficient band width to detect a 30 μsec burst from the pulsed plasma thruster can range to 10^{-9} A/cm^2. Ion current release during the burst is almost at kiloampere levels, hence, the Faraday cup sensitivity is at the $\varepsilon \sim 10^{-12}$ cm^{-2} level, which is well below even the most severe integration requirements. Furthermore, ion cup collectors are significantly less subject to error from thin layers of contaminant films than are probes for electron detection, such as Langmuir probes. For this reason, the Faraday cups will not require self-cleaning mechanisms. The Langmir probe recommended is, on the other hand, a directly heated, self-cleaning probe.

Sufficient bandwidth exists for both the Langmuir probe and Faraday cup so that measurements can be taken during time-of-flight of the plasma burst across the testing chamber and, hence, in the period before the burst encounters facility walls. This is the most effective way to provide for efflux measurements to very low ε levels without the effects of wall generated or wall scattered fluxes.

In summary, spacecraft surfaces should avoid the energetic plasma core which extends out to 40 degrees from the centerline of the thruster in the direction of the electrodes, and out to 25 degrees in the direction

of the fuel bars. Surfaces in this region will be subject to sputter erosion. Sensitive surfaces are best protected by placing them at large divergence angles from the thruster centerline, and at as large a distance as possible to enable efflux dilution (it roughly dilutes as the square of the distance). If necessary, beam shields should be placed on the thruster to protect sensitive surfaces. In all cases, efflux measurements must be taken for interactive effects analyses, which typically proceed to the $\epsilon = 10^{-9}$ cm^{-2} level. Measurements should be taken with the beam shield installed if it is to be used operationally.

4.2 ELECTROMAGNETIC COMPATIBILITY

The pulsed plasma thruster relies upon an intense burst of electromagnetic energy to impart reactive impulse to a spacecraft and hence, is inherently a source of radio frequency (RF) noise. Careful attention to electromagnetic compatibility of pulsed plasma thrusters with the spacecraft bus and payload in the past has resulted in successful integration of smaller versions -- much less than 1 millipound average thrust -- of these thrusters on a variety of programs,[23, 24, 25, 26] several of which have operational flight experience.

Electromagnetic compatibility (EMC) tests of the 1 mlb pulsed plasma thruster operating in air were made with an unshielded test set.[27] Test results exceeded MIL-STD-461 and MIL-STD-1541 radiated environment limits, but the fact that MIL-STD limits are exceeded does not mean that there is an incompatibility with a satellite. Many circuits can withstand environments well above MILL-STD limits. The actual levels, however, that circuits can tolerate are determined by analysis and/or test. The approach

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[23] W. J. Guman and D. M. Nathanson, "Pulsed Plasma Microthruster Propulsion System for Synchronous Orbit Satellite, " Journal of Spacecraft and Rockets, Vol. 7, No. 4, April 1970, pp. 409-415.

[24] W. J. Guman, "Task 1 - Design Analysis Report, Pulsed Plasma Solid Propellant Microthruster for the Synchronous Meteorological Satellite," NASA-CR-122358, December 1971.

[25] K. I. Thomassen, "Radiation from Pulsed Electric Thrusters," Journal of Spacecraft and Rockets, Vol. 10, No. 10, October 1973, pp. 679-680.

[26] W. J. Guman and S. J. Kowal, "Pulsed Plasma Propulsion System for TIP-II Satellite," JANNAF Propulsion Conference, 1975.

that should be taken to assure successful integration of the flight configured thruster is to perform a computerized EMC analysis as is outlined below.

The purpose of computerized EMC analysis is to analytically determine the effects of radio-frequency fields generated by the thruster. Existing computer models, that can handle transient interference and threshold devices, may be used for this purpose. For example, a SEMCAP (Specification and Electromagnetic Compatibility Analysis Program) model of the DSP spacecraft described in Section 3.5 was developed and implemented for predicting EMC response of that spacecraft. Basically, SEMCAP resorts to a computerized analysis because of the huge numbers of terminal-to-terminal wiring involved in any spacecraft system. After all of the system descriptions such as the source and receptor characteristics and the wiring layout are put into the computer, the coupling is computed via four types of coupling matrices. Fortunately, since many of the wires run in common bundles or cable harnesses, the size of the matrices is not comparable to the number of terminals. The outputs of SEMCAP are given in several ways:

- Voltages at each receptor terminal
- Margins of immunity in dB
- Alphabetical indicators of negative immunity margins

The additions to SEMCAP to model electromagnetic interference (EMI) from the pulsed plasma thruster are small in comparison to the original effort required to implement SEMCAP. After the pulsed plasma thruster EMI is modeled, a SEMCAP analysis will identify those circuits having an EMC safety margin less than +6 dB. In addition to identifying these circuits, their exact safety margin will be computed.

Following the intitial analysis described above, the thruster EMI model can be modified to achieve +6 dB safety margin. These data can then be used to specify the EMI requirements for the thruster if it is to be compatible with the spacecraft system.

[27] M. Begun and W.J. Guman, "Pulsed Plasma Radio Frequencey Interference Studies," AFRPL-TR-77-85, September 1977.

4.3 INTERACTION WITH COMMUNICATIONS

The earth facing side of typical communications satellites contains a family of antennae for up-link and down-link RF signals. Under typical operating conditions, the RF lines-of-sight are approximately perpendicular to the thrust axes and, hence, the RF signals are not required to traverse the comparatively dense plasma regions in the thruster's plume. It should be noted, however, that S-band signals will be completely absorbed for a distance of ~ 20 meters along the thrust axis during the period of the plasma burst (ρ_+ diminishes to levels of $\sim 10^{11}$ ions/cm^3 at ~ 20 meters from the thruster) and that appreciable RF refraction effects of S-band signals will exist for distances of almost 100 meters from the thruster.

Several factors combine to diminish the RF refraction effects of the plume. The first of these is the use of higher frequencies (such as X band) where the plume is more transparent to the RF and both the critical absorption zone and the critical refraction zones in the plumes shrink significantly. The second factor is the wide angular separation between the RF lines-of-sight and the thrust axes, so that only the widely divergent portions of the plasma plume will move in the direction of the RF transmission path. A third factor is the shielding effect of the spacecraft body. This shielding acts to further diminish the plume density along the RF lines-of-sight.

In order to determine whether or not significant levels of RF refraction and signal loss occur during an impulse burst, thruster tests should include the live linkage of a representative communications antenna to a transmitting and/or receiving antenna to observe possible RF signal dropout during the burst period. These tests would also answer any concern about possible initiation of microwave breakdown resulting from high electron density in plasma regions near the antenna.

The potential impact of the pulsed plasma thruster on DSCS-III communication channels has been examined. While it is difficult to assess the effects during a 30 μsec pulse, it is possible to make some worst case analyses, which assume complete interruption during the 30 μsec burst. In this case, it was shown that there are satellite and/or ground station system level actions which can be implemented to avoid serious effects on the communication services.

5. RELIABILITY AND REDUNDANCY

5.1 FLIGHT EXPERIENCE

There have been four fully flight-qualified microthruster sybsystems developed since Teflon solid propellant pulsed plasma propulsion entered development in 1966. The operating characteristics of these four subsystems are summarized int Table 18.[28] Two of these microthruster subsystems, the LES-6 and the TIP-2,-3, have proven flight experience. The LES-6 subsytem was flown in 1968[29] and accumulated over 8900 hours of operational experience performing east-west stationkeeping. The Air Force/MIT Lincoln Laboratory LES-6 was a spin-stabilized experimental communications satellite. The propulsion subsystem was part of an autonomous stationkeeping control system and kept the satellite within the design value of the deadband.

Table 18. Flight-Qualified Pulsed Plasma Microthruster Propulsion Subsystem[28]

	LES-6	SMS	LES-9	TIP-2,3
Impulse bit				
(μN·s)	26.7	111	307	400
(μlb·s)	6.0	25	69	90
Equivalent Steady Thrust				
(μN)	17.8	89-200	307-1840	400 max
(μlb)	4.0	20-45	69-414	90
Bus Power (W)	2.5	8.7-14.6	25-150	30 max
Repetition Rate (s^{-1})	0.16	0.83-1.83	1-6	single pulse/1
Total Number of Pulses	$1.2(10^7)$	$1.6(10^7)$	$3.4(10^7)$	$6(10^6)$
Total Impulse				
(N·s)	285	1780	11,800	2450
(lb·s)	64	400	2650	550

[28] B.A. Free, W.J. Guman, B.G. Herron, and S. Zafron, "Electric Propulsion for Communications Satellites," A AA 78-537, April 1978.

[29] A. Braga-Illa, "The Future of Self-Contained Control of Synchronous Orbits," AIAA 70-479, April 1970.

The TIP-2 propulsion subsystem was launched in October 1975 but due to satellite problems was not activated until January 1977. The two thrusters aboard the Navy-Johns Hopkins University APL TIP-2 navigational satellite as well as the TIP-3 satellite, launched in September 1976, were designed to keep the gravity gradient stabilized satellites in a drag-free state as part of a closed loop autonomous control system.

5.2 RELIABILITY ASSESSMENT

Preliminary propulsion subsystem reliabilty assessments may be made using the failuure rate data for the 1-millipound thruster subsystem equipment listed in Table 19. The failure rate for the power conditioner was determined from analysis of preliminary circuit diagrams with anticipated use of flight quality components in accordance with MIL-HDBK-217B, Notice 2. The gimbal assembly data were obtained from current design experience on the TRDS (Tracking and Data Relay Satellite) program. The propellant-discharge assembly represents the area of greatest uncertainty assessment. Within this assembly, high reliability energy storage capacitors of the appropriate size for the millipound thruster are presently under development, but definitive failure rate data for them are currently unavailable. Accordingly, preliminary efforts were undertaken to define the anticipated range of failure rates for this assembly. From this effort, an anticipated range from 500 to 3500 bits was established.

Table 19. Active Equipment Failure Rates

$\lambda_{\text{Power Conditioner}}$	= 990	From preliminary MIL-HDBK-217B analysis
$\lambda_{\text{Gimal Drive Electronics}}$	= 684	TDRS* data
$\lambda_{\text{Gimbal Drive Stepper Motor}}$	= 143	TDRS data
$\lambda_{\text{Propellant-Discharge Assembly}}$	= 500 to 3500	Anticipated range of applicability

1 bit = 1 failure/10^9 hours

*Tracking and Data Relay Satellite System

The data in Table 19 represent active failure rates, i.e., applicable when the equipment is in use. Recommended standby (inactive) rates are $\lambda_{standby} = 0.1\ \lambda_{active}$ for the electronics and gimbal assembly. The standby rate for the propellant-discharge assembly depends on the influence of the propellant feed system, which is active within that assembly all the time. For initial analysis, $\lambda_{standby} = 0.5\ \lambda_{active}$ for the propellant-discharge assembly assumes that the feed system and energy storage capacitor failure rates are of similar magnitude.

5.3 REDUNDANCY CONSIDERATIONS

Redundant thrusters may be employed to reduce the sensitivity of propulsion subsystem reliability to the propellant-discharge assembly failure rate. In this case, it is appropriate to note that a single power conditioner, with minor modification, may be used to operate either of two propellant-discharge assemblies.

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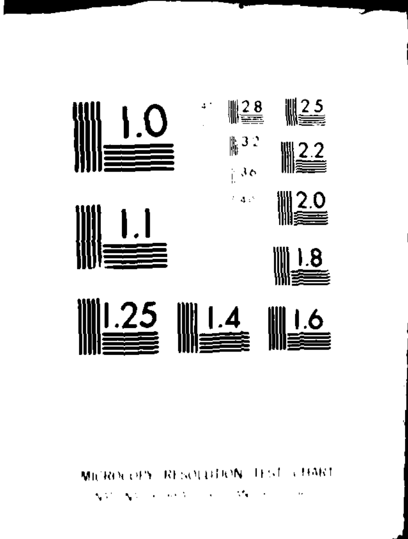

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